MOM•E

MOON•ORBITING MOTHERSHIP EXPLORER

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1.0 Introduction

The National Aeronautics and Space Administration proposed that a new class of robotic space missions and spacecrafts be introduced to “ensure that future missions are safe, sustainable and affordable”. Indeed, the United States space program aims for a return to manned space missions beyond Earth orbit, and robotic explorers are intended to pave the way. This vision requires that all future missions become less costly, provide a sustainable business plan, and increase in safety.

Over the course of several fast feasibility studies that considered the 3 drivers above, the small-scale, consumer-driven Moon-Orbiting Mothership Explorer (MOM-E) mission was born. MOM-E’s goals are to enable space exploration by offering a scaled down platform which carries multiple small space explorers to the Moon. Each payload will be dropped at their desired destination, offering a competitive price to customers. MOM-E’s current scope of operations is limited to the Moon and will be used as a proof of concept mission. However, MOM-E is specifically designed with the idea that the platform is scalable.

2.0 Requirements

These requirements outline the overall mission architecture, functionality and operations:

- SYS-001: The spacecraft shall support at least 2 lander missions.
- SYS-002: The spacecraft shall transport payloads from Low Earth Orbit (LEO) to a circular Lunar orbit.
- SYS-003: The spacecraft shall serve as a communications relay for payload.
- FNC-SYS-001: The spacecraft shall support a Google Lunar X-Prize mission.
- FNC-SYS-002: The spacecraft shall communicate with Earth.
- FNC-SYS-003: The spacecraft shall take health data checks throughout mission lifetime.
- FNC-SYS-004: The spacecraft shall release and/or activate payloads upon reaching Lunar orbit.
- FNC-SYS-005: The spacecraft shall release and/or activate payloads upon reaching circular Lunar orbit.
- OPR-SYS-001: The spacecraft shall comply with all applicable FAA and FCC regulations.
- OPR-SYS-002: The spacecraft shall support full operations for a minimum of 3 years.
- OPR-SYS-003: The spacecraft shall support deployable and non-deployable payloads.
- OPR-SYS-004: The spacecraft shall withstand environmental conditions.

See Appendix A for all requirements.

3.0 Concept of Operations

MOM-E will undergo a multitude of operations throughout its lifetime. Initially, it will be launched into LEO, where all subsystems will be dormant. Once in LEO, the spacecraft will power on and perform health checks on the payloads while station keeping. The spacecraft's kickstage will perform a burn, transferring the spacecraft from LEO to Moon orbit. During this transfer, the communications subsystem will be relaying position and orientation data back to Earth, while all other subsystems will be in a low-power mode.

As the spacecraft nears the Moon, the kickstage will perform another burn to circularize its orbit. After burn completion, the kickstage will depart from the spacecraft, and every subsystem will prepare the spacecraft for deployment of the payloads by performing health checks, de-spinning/stabilization, and high-gain antenna positioning. The payloads will be deployed or initiated and will then begin their individual missions. MOM-E will continue to orbit the Moon for 3 years, serving as a communications relay for the payloads. After 3 years, MOM-E will shut down and naturally de-orbit. If adequate power permits, ADCS will assist the de-orbiting process by thrusting the spacecraft towards the Lunar surface.

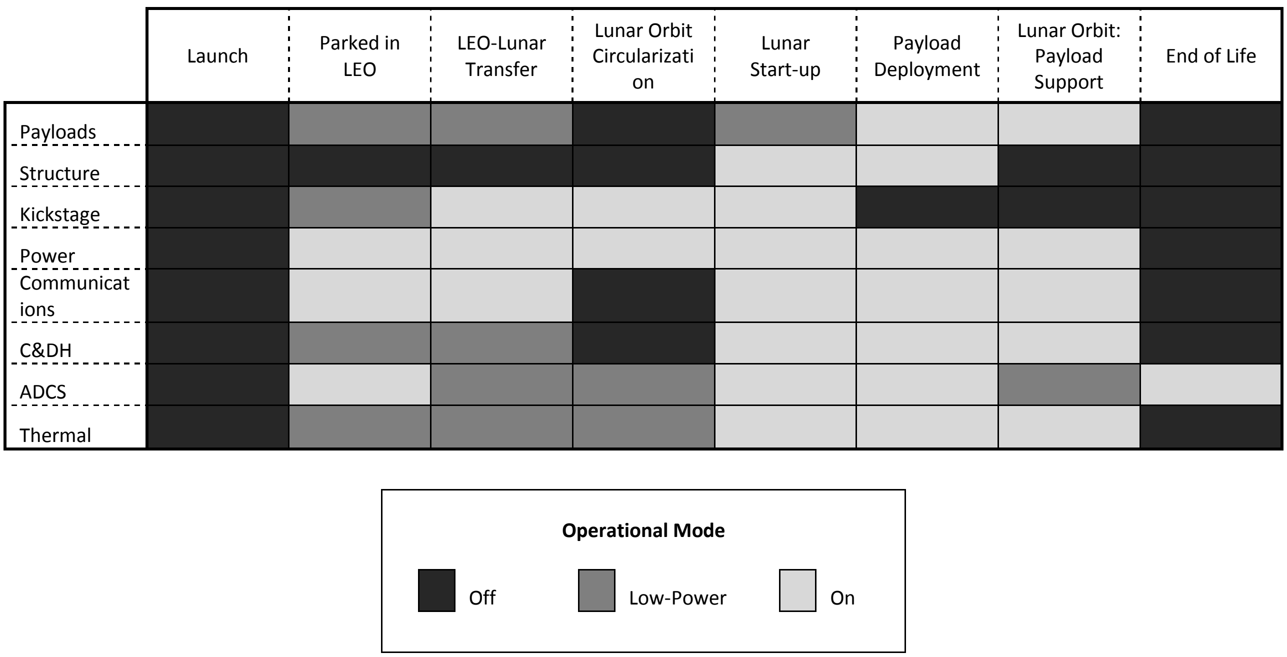


Table 1: Subsystem Operation Modes throughout MOM-E Mission Lifetime

4.0 Subsystem Details

MOM-E has an architecture that requires the integration of payload, structures, power, thermal, radiation, trajectory, propulsion, attitude determination and control, command and data handling, and communications subsystems.

4.1 Payload

The team studied previous Lunar and Martian lander and orbiter missions and future Lunar missions to determine payload bay size. A comparison can be seen between the Martian missions Spirit and Opportunity, the Google Lunar X-Prize competitor Team Frednet's rover, and the MOM-E maximum individual payload capacity:

Mission	Mass (kg)	Volume (m^3)	Data Rate (kbps)
Team Frednet	150	0.568	9
Spirit/Opportunity	820	5.3	256
MOM-E	500	3.50	2500+

Table 2: Lander Comparison

See Appendix B for additional heritage.

The total feasible payload mass and volume was 1500 kg and 10.5 m^3, which would be more than enough for Spirit and Opportunity. For business sustainability purposes, this amount would be divided into multiple bays to accommodate multiple customers – including those with payload types such as orbiters and CubeSats, which are typically smaller than landers.

As the purpose of MOM-E is to shuttle and support a minimum of 2 landers/rovers to Lunar orbit, the following requirements have been established:

- Subsystem shall support small, medium, and large payloads with a standard interface.
- Subsystem shall have a maximum mass of 500 kg, volume of 3.5 m^3 (1.8 m diameter and 1.4 m height, justified by heritage studies).
- Subsystem shall use an ESPA ring to connect secondary payloads around a larger primary payload.
- Subsystem shall use a Mark II Lightband interface as a connection between the ESPA ring and payload, ejecting payloads via hitch/spring mechanism (Appendix C).

From these requirements, the team designed the subsystem such that the IT uses only the port dimensions of the ESPA ring carved into a flat plate, and the ESPA ring's umbilical cord technology to electrically connect the payload to the mothership.

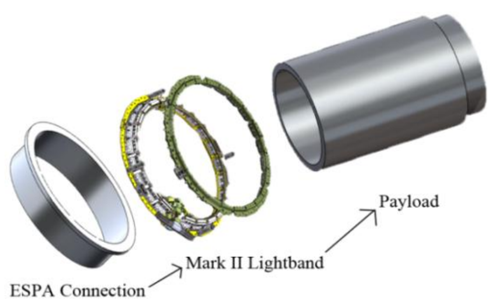


Figure 1: Payload Connection Method

Landing on the Moon requires a large change in velocity (and consequently, a large amount of fuel). The team determined that for a specific impulse of 320 s, a 500 kg payload uses about half its mass in propellant to slow down and land, leaving over 200 kg to conduct scientific research. For details on this calculation, see

Appendix D.

The payloads themselves will be designed and fabricated by outside sources, but they must adhere to specific requirements laid out in the MOM-E Payload User's Manual (Appendix E).

4.2 Structures

The MOM-E structure weighs approximately 1,000 kg with contingency (Appendix F). It is comprised of the following components, all made of Aluminum 7075 (Appendix G):

- **Modules:** The modules are designed to house the components that support the mothership operations. The modules have variable heights for scalability. Externally attached are the mothership's antennas, solar panels, and sensors.
- **Payload Bays:** The payload bays have variable heights to accommodate customers' specific payload types, which include landers, satellites, CubeSats, etc.
- **Solar Panel Gimbals:** The solar panels are attached to the top module by the gimbals, which allow for the panels to rotate about 2 axes to maximize power generation.
- **Lightband:** The payloads are attached to the top of the payload bays by the Lightbands discussed in the Payload section.
- **Lock-and-Spring Adapters:** These adapters (similar to the 1664HP-1000 separation system developed by KSC) provide a 3-point hard mount that can be reliably separated from the mothership. The kickstage /mothership interface and the payload interface are attached using these adapters (Appendix H).

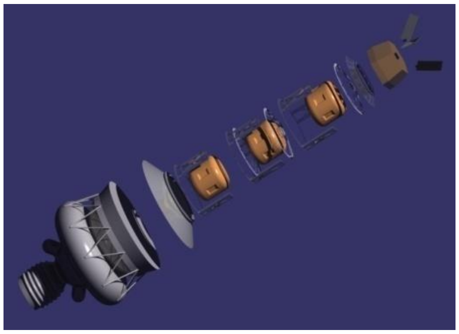


Figure 2: MOM-E Structure

For more detail on the structure layout, see Appendix I.

Mothership Deployment Operations: The mothership deploys sections from the bottom to the top, using the following process:

1. Release kickstage once in Lunar orbit.
2. Eject payload bay surround bottom payload.
3. Release payload from Lightband.
4. Repeat Steps 2 and 3 for remaining payloads.

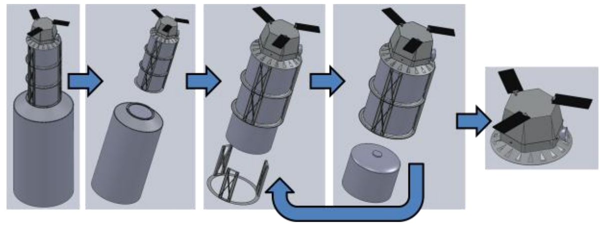


Figure 3: Separation Process for MOM-E

Stress Analysis: In order to ensure that MOM-E will survive the 5g load during launch, stress analysis was performed on the components, and they were modified to meet the requirement of a minimum safety factor of 1.25 while remaining within the mass budget. The evaluated safety factor for the structural components is shown, and the detailed results are in Appendix J**.**

Structural Component	Evaluated Safety Factor
Top Module	3.19
Bottom Module	2.24
Module Adaptor Plate	2.15
Payload Plate	1.38
Bottom Trusses	3.46

Table 3: Minimum Safety Factors of Structural Components from ANSYS Simulation

4.3 Power

The power subsystem gathers, manages, allocates, and distributes power to all other subsystems on the spacecraft. Large-scale missions that target deep space would consider using a radioisotope thermoelectric generator; however, the scarcity of the fuel used in an RTG and the costs associated with its attainment makes this source infeasible. As a result, solar power has been chosen for the mission. Table 4 details the components of the power system:

Quantity	Description	Function	Mass (kg)
3	*Solar Arrays* (Solar Cells + Supporting Hardware)	Power collection from the space environment	4.33 /array
1	*Power Management and Distribution Unit (PMAD)*	Distribution and Regulation of power to all subsystems	8
60	*Primary/Deployment Batteries* (Lithium Sulfur Dioxide)	Power for one-time Mark II Lightband and rail deployment	0.063/cell
20	*Secondary Batteries* (Lithium Ion)	Power source for Eclipse operations	1.1/cell
1	*Wiring/Harnessing*	Power lines for subsystems and cable management	20 (max)
3	*Solar Array Gimbals*	Solar array deployment and gimbaling mechanism	1.4/gimbal

Table 4: Components of Power System

Solar Array: The solar cells are composed of a triple junction GaInP/GaAs/Ge composite with an efficiency of 28%. They will be mounted on 3 panels, each with an area of 0.39 m^2, providing a beginning-of-life (BOL) power of 450 W and an end-of-life (EOL) power of 413 W. Two panels are designed to provide 100 W of power continuously, both in sun and eclipse times. The other is for redundancy. Excess power generated will be stored in secondary batteries.

Power Management and Distribution (PMAD): The PMAD distributes power to individual components and ensures safety by preventing component overcharging. Four components make up the PMAD.

1. *Array Power Regulation Module*: This module processes the power from the array to the main bus of the distribution module. Excessive power from the array will be mitigated in this unit via shunt resistor.
2. *Battery Charge / Discharge Regulation Module*: To ensure proper battery health is maintained, this module prevents overcharging and excessive discharging.
3. *Power Distribution Module/ Command and Monitoring Module*: This module can be considered part of C&DH or independent module inside power system. It monitors the

condition of the power system, so it protects s the system. The power distributed as commanded from C&DH or telemetry.

4. *DC – DC Converter*: Since various components require various voltage inputs, the output voltages from the PMAD are regulated for each specific component. Due to power loss that occurs when transferring power over long distances, voltages are also boosted in this module to account for losses in wires.

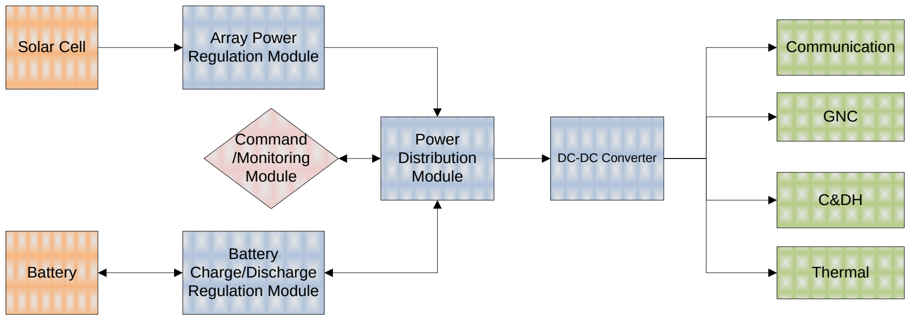


Figure 4: PMAD System

Deployment Batteries: Independent primary battery strings will interface with the deployment mechanism and upon command from the mothership flight computer, these batteries will provide the one-time power to enable the payload deployment. Two sets of sulfur dioxide batteries will provide both the activation string and a redundant string in case of failure. Sulfur dioxide batteries were chosen due to a long shelf life with small discharge, low temperature operating range, and high energy density.

Wire Harnesses: These provide a communication link between the flight computer and the payloads to enable system health verifications and command the subsystems. They will be housed inside not one, but 2 radiation-hardened rails of each payload bay for redundancy.

Modes of Operation and Associated Power and Energy Budgets: Normal and secondary modes of operation have been created from the requirements of the mission. The normal modes are modeled around communications, hibernation (power-saving), and ADCS. The secondary modes are modeled around payload deployment, stabilization post-payload deployment, and the transit from the Earth to the Moon. Each mode incorporates the requirements of each system, and a worst-case eclipse scenario was modeled. In all scenarios, the depth-of-discharge of the secondary batteries was less than the power available to charge the batteries in a post-sun pass. Appendix K and Appendix L detail the power and energy budgets for the modes.

4.4 Thermal and Radiation

Thermal Design: The spacecraft undergoes a large temperature range in LEO, transfer to the Moon, and Lunar orbit. In LEO, the ship will be heated by the Earth's atmosphere and the Sun. In transit, the ship will be in the Sun constantly. In Lunar orbit, the ship will have to handle the changes from light to dark.

To determine the thermal conditions, the team made a model of the thermal environment. Based on this, the module does not get overly hot when traveling from the Earth to the Moon.

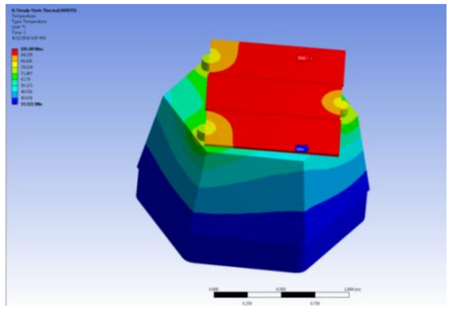


Figure 5: Transit Thermal Conditions, 30⁰ Lighting with Rotation

To mitigate any excessive heating that does occur, the ship will spin along its secondary access to ensure an even distribution of heat. Additionally, the spacecraft has radiators and a layer of Multi Layer Insulation (MLI) to help keep the interior of the spacecraft from heating up. These are useful in Lunar orbit, also, as they helps keep the interior from changing temperature as the ship moves between sunlight and eclipse. See Appendix M for further details.

Radiation Design: The mission exposes MOM-E to a wide range of radiation. The ship will have to endure radiation in LEO, in the Van Allen belts, and outside the magnetosphere. LEO radiation is not a concern, as the radiation is only slightly elevated from the levels on Earth. The Van Allen belts will impart a large amount of radiation in the ship; however, the spacecraft will not be in them for long. Outside of the magnetosphere, though, MOM-E will be exposed to the full force of the solar wind, coronal mass ejections, and galactic cosmic rays (GCRs).The Earth Moon Mars Radiation Environment Model, SOHO, and GOES were used to determine the expected total ionizing dose (TID) and worst case acute dose in this environment.

The ship's structure and thermal system provide insufficient radiation protection. As such, the team has added an extra layer of high Z non-structural material to the module. Additionally, the sensitive components have been encased in RADPAKs. All of the shielding should provide the ship with enough protection to function for the expected 3 years in solar maximum (higher TID) or minimum (more GCRs, or more single event effects). See Appendix N for further details.

4.5 Trajectory and Propulsion

MOM-E is required to be in Lunar orbit. Due to this, an appropriate launch vehicle (LV), kickstage, and transfer orbit is necessary. The LV must be able to move the MOM-E payload, kickstage, and enough fuel for Lunar transfer insertion to LEO. The kickstage must be capable of multiple burns and provide the power to send the MOM-E payload to the low Lunar orbit (LLO). The transfer orbit must use as little fuel as possible and have a short time of flight (TOF).

Though an analysis was done specifically for Lunar mission LVs, kickstages, and transfer orbits, this concept is scalable to other robotic missions within the solar system.

To accomplish the requirement of Lunar orbit, the Proton-M rocket, the Block DM-2M kickstage, and the direct elliptical transfer orbit were selected. The Proton-M is able to get 22,000 kg to LEO. This 22,000 kg payload is enough to include the spacecraft, the Block DM-2M kickstage, and the necessary fuel to achieve the direct elliptical transfer. The Block DM-2M is able to get the spacecraft, a mass of 3,350 kg, to LLO with excess fuel for inclination changes and can execute multiple burns. This not only satisfies the requirement of achieving LLO, but it improves upon the requirement by giving the MOM-E spacecraft the ability to increase or decrease the orbital inclination by 18^o from the selected inclination of 10^o. This provides the MOM-E spacecraft with a range of orbital inclination of -8 to 28^o. This provides more options for the customer and has the potential to draw new customers.

The direct elliptical transfer orbit requires 4 km/s of ΔV and has a TOF of 5.2 days. The 4 km/s of ΔV is an achievable amount using the selected kickstage and LV. When considering radiation, thermal, and power constraints, a transit time of 5.2 days should not produce problems. See Appendix O for calculations.

4.6 Attitude Determination and Control

The Attitude Determination and Control Subsystem (ADCS) is responsible for the spacecraft-orienting maneuvers. These maneuvers have intrinsic requirements set by other subsystems to both facilitate various parts of the mission and to meet specified mission objectives. The requirements mainly pertain to pointing and stabilization, and are defined as follows:

- The ADCS shall maintain the desired orbit altitude and inclination through the mission.
- The ADCS shall maintain orientation and pointing for proper communications with Earth and the payloads when necessary.
- The ADCS shall orient the spacecraft to allow for proper deployment of payloads.

At minimum, the system must be composed of attitude sensors, navigation sensors, and control actuators acting in conjunction to translate the attitude and location into controlled operations.

Mission Phases: ADCS has 4 phases throughout the mission:

1. LEO Systems Check phase, which begins after the spacecraft is released from the launch vehicle and will culminate with the activation of the kickstage and spacecraft ADCS
2. Journey from Earth to Lunar Orbit phase, in which the spacecraft operates with the kickstage's internal ADCS and spins about its vertical axis to eliminate thermal issues
3. Payload Insertion Maneuvers phase, in which responsibilities are passed off to the spacecraft's internal ADCS and the system must maintain its proper orbit and stability while ejecting nearly half of the initial mass of the spacecraft in separate payloads
4. Payload Support Orbiter phase, in which attitude control is primarily driven by communication pointing requirements

Sensors: Several sensors are integrated onboard the spacecraft in order to achieve redundancy in case of sensor failure. A pointing accuracy of 1.28 ^{o}, as required by the communications subsystem, drives the pointing precision for all sensors since communication with the Earth must be possible. The most accurate sensors are the 2 star trackers, which are accurate +/- 90 arcseconds and can track up to 10 ^{o}/s. The 2 sun sensors on the spacecraft have accuracies of +/- 1^o, and are mainly included for redundancy. The inertial measurement unit (IMU) has a tri-axis gyroscope and accelerometer, making it an important addition to the sensor design. The IMU, unlike the other sensors, does not require tracking of external sources, and can track between 75-300 ^{o}/s (depending on axis), which is significantly better than the star trackers. Combined, these sensors provide adequate accuracy and tracking capabilities.

Tracking: The spacecraft's position and velocity will be determined using data provided by the communications transponder. The distance between the ground station and the spacecraft can be determined by measuring the downlink propagation time. Similarly, the spacecraft's velocity can be determined by measuring the Doppler offset of the signals received from the spacecraft. These procedures have been proven to provide accurate measurements of a spacecraft's orbital position and velocity while the spacecraft is in contact with the ground station.

Control Actuators: The control system – which is responsible for reorienting and stabilizing the spacecraft – consists of 4 reaction wheels and 10 monopropellant hydrazine thrusters, allowing for full attitude control and providing redundancy in the case of component failure. Due to their relatively small size in relation to the entire spacecraft, the reaction wheels (which provide 5 mN-m/wheel) can be used for both fine pointing and mitigating small disturbance torques.

The thrusters will expel the excess momentum stored in the wheels and will provide station-keeping, pointing and spin-stabilizing capabilities for the spacecraft. Different combinations of active thrusters can be utilized to yield 6 degrees of freedom.

Thruster Integration: Since the payloads will completely detach from the spacecraft, it is not feasible to mount any thrusters to them. Instead, they will be positioned in 2 tripod formations, one either end of the spacecraft, with each thruster spaced 120° apart and angled 45° off of the vertical axis to provide torque on different axes. Four thrusters will be placed on opposite sides of the spacecraft to provide roll moments.

Appendix P contains a CAD model of this layout.

4.7 Command and Data Handling (C&DH)

C&DH serves 4 primary functions: command processing, telemetry housekeeping, payload data processing, and data storage and protection. The C&DH subsystem is the central computer aboard the spacecraft used to interpret, process, and execute MOM-E's internal and ground commands. Telemetry information arising from nominal housekeeping operations, watchdog functions to ensure proper functionality of all electronic components, and data storage from payload operations are all run via the C&DH subsystem.

Flight Components: The subsystem is comprised of a central processing unit (CPU), a watchdog processor unit, the system memory, the system main memory or hard drive, the memory bus, input/output (I/O) bus, and I/O unit. The CPU is responsible for all processing and validation operations. The system memory and random-access memory (RAM) contains all functional programs and software applications. The main memory stores all the scientific data obtained from the payloads and sensor packages. The memory bus is the wiring to transmit data to and from the processor, and the I/O bus is the transmission wiring between the CPU and the various subsystems. The I/O controllers control the traffic on the I/O bus.

Below is a detailed explanation for each main flight component:

- **Main System Processor:** The Proton 400k-L processor was chosen as the MOM-E's primary CPU. The processor is radiation-hardened and compatible with UART, SPI, CAN, and I^2C protocols. It can easily interface with the various ADCS sensors and is bit-flip adverse. More information on the processor can be found in Appendix Q.
- **Main Memory:** The mission will employ the use of Solid State Discs (SSD) as the main system memory. The mission requirements dictate that the C&DH subsystem is able to retain at least 500 GB of data. C&DH has decided to create a storage device that consists of 8-stacked MTRON PRO 7500 128 GB units. The MTRON 7500 has flight heritage aboard the International Space Station and features space ready components. Primarily, the SSD has radiation resistant components and strong vibrational endurance.
- **Data Bus:** The Ethernet protocol was chosen for the main system bus for 2 primary reasons: user base and speed. The Ethernet protocol is well documented and easy to use. There are a number of existing tools that will allow users to interface with the bus. Ethernet is also fast (100 Mb/s), which will allow the flight computer to send all the data necessary to the communications system to utilize the full 25 Mb/s downlink. Further information on the data bus trade study can be found in Appendix R.

Architecture: Figure 6 shows the architecture of the subsystem. These connections are governed by the data rates of the devices and the need for redundancy.

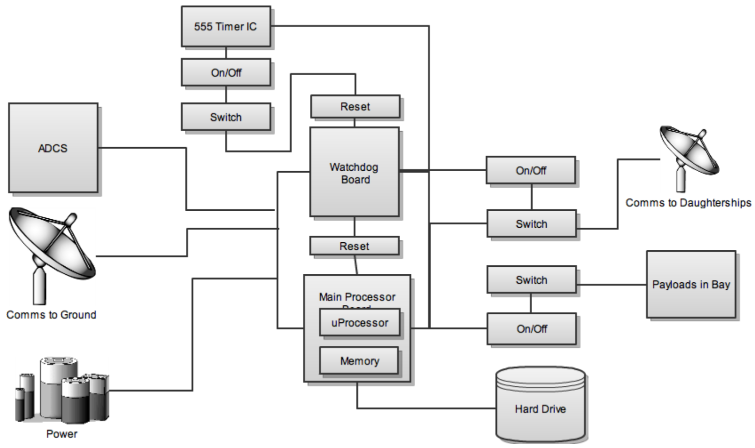


Figure 6: High-Level Schematic of C&DH Architecture

Wiring Harness: Figure 7 shows the high-level function diagram of the wiring between C&DH in relation to the other subsystems. The redundant lines create a more fault-resistant harness.

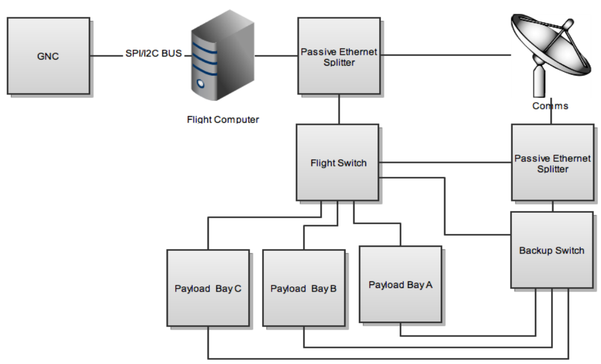


Figure 7: Fault-Resistant Ethernet Ring on MOM-E

This wiring scheme allows for redundancy in payload connections and the communications relay. The Ethernet switches control the traffic on the network and can be used to route data around broken connections. Ethernet pin/pad connections are used between the payloads and the rest of the harness so the payloads can be easily released from the mothership.

Operational Modes: Operations will follow a procedure outlined in Figure 8*:*

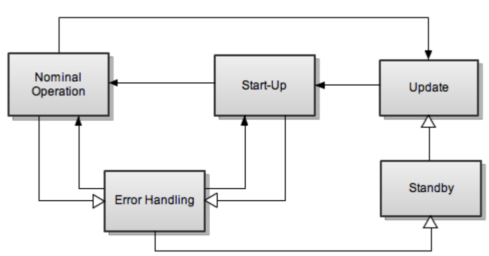


Figure 8: Operational Modes

Normal spacecraft operations repeat the progression Start-Up, Nominal Operation, Update, as shown by the black arrows. If an error is perceived during health checks, Error Handling mode is entered. During this mode, the system will attempt to determine if the source that triggered the error was an anomaly or if the system is indeed outside the operational limits. If the system is determined to be outside the operational limits, it will enter standby/safe mode. Otherwise, the error handling process will terminate and the system will continue in its current mode.

4.8 Communications

The communication subsystem acts as a communications relay for payloads. Additionally, the system must be capable of transmitting telemetry back to Earth. The design drivers of this system come from a 3 sources: the mission objective, ground station selection, and other MOM-E subsystems. The baseline MOM-E communication design accommodates high data rate transmission to Earth and provides omnidirectional coverage for payload communication.

MOM-E's communication architecture, shown in Figure 9, consists of 2 transceiver units connected to 1 high-gain parabolic dish antenna (HGA), 3 low-gain dual band patch antennas (LGAs), and 1 low-gain micro-patch array. Each transceiver unit has 1 General Dynamics multi mode S-Band transceiver, 1 custom L3 S-Band receiver, and 1 General Dynamic X-Band transceiver. With this design, either unit can control all S-Band or all X-Band communication. The X-Band transceiver transmits data and telemetry down to Earth and receives commands from the ground. To enable high downlink data rate, a Traveling Wave Tube Amplifier (TWTA) is connected to the X-Band transceivers.

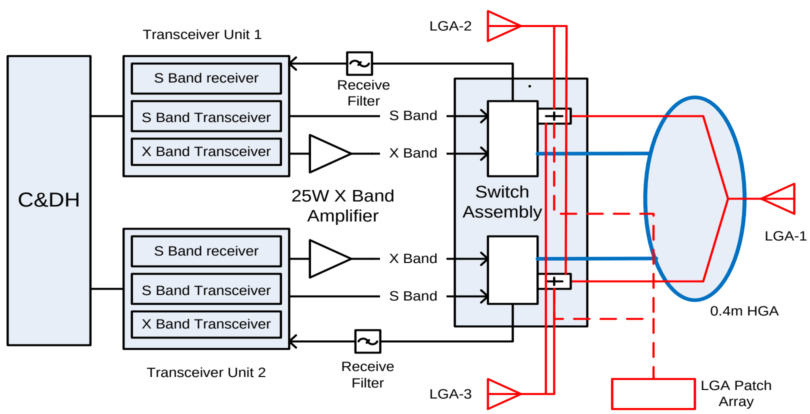


Figure 9: Communication Architecture

The design fulfills MOM-E's requirement to act as a communications relay. Each transceiver unit is responsible for either Earth communication or payload communication. To add redundancy to the system, either unit can take complete control over MOM-E's communication, at a reduced capacity, should one fail. An S-Band receiver is included due to the limited receive capability of the S-Band transceiver. A switch assembly manages the transmit-and-receive signal flow for the system.

MOM-E has 4 options for selecting a ground station: NASA's Deep Space Network (DSN), a commercial ground station network, an in-house ground station, or some combination of these. The ground station selection directly affects the possible frequencies MOM-E can use, the amount of coverage time available, and the mission cost.

	Bands Supported	Dish Size (m)	Gain (dBi)	S/C Dish (X-band) (m)	Location	Startup Cost (USD)	Operational Cost/Year (USD)
Deep Space Network	S, X, Ka	34	56.8 (S), 68.3 (X), 79 (Ka)	0.4	Global	0	1,000,000
Universal Space Network	S, Limited X, Ku , Limited K, Limited Ka	10	45(S) 58(X)	0.9	Global	0	400,000
In-House System	S, X	5.5	65 (S)	N/A	Ann Arbor, MI	2,000,000	350,000

Table 5 compares the options of DSN, a commercial ground station network (Universal Space Network), and an in-house system. The 2 differences to note are the required dish size and location. The ground station dish size drives the antenna size on MOM-E (Appendix S). A smaller ground station dish increases spacecraft mass and decreases available payload mass, which reduces the profit for the MOM-E mission. The ground station location determines the complexity of MOM-E's operational modes. Additionally, the scalability of the ground station capabilities is important. Higher frequencies, like X-band, allow for high data rates at the Moon and beyond. Based on the trade studies, DSN is reasonably priced and provides ample capabilities for the first 3-4 years (first mission).

	Bands Supported	Dish Size (m)	Gain (dBi)	S/C Dish (X-band) (m)	Location	Startup Cost (USD)	Operational Cost/Year (USD)
Deep Space Network	S, X, Ka	34	56.8 (S), 68.3 (X), 79 (Ka)	0.4	Global	0	1,000,000
Universal Space Network	S, Limited X, Ku , Limited K, Limited Ka	10	45(S) 58(X)	0.9	Global	0	400,000
In-House System	S, X	5.5	65 (S)	N/A	Ann Arbor, MI	2,000,000	350,000

Table 5: Ground Station Trade Study

MOM-E is capable of downlinking 13 Mbps to Earth with the HGA over X-Band at 28W input power to a 34 m diameter dish at DSN ground stations. The same X-Band transceiver is capable of receiving 4 kbps of commands to from Earth. The patch antennas are capable of transmitting up to 6 Mbps to the payloads over S-band, but 11W input power achieves the 4 kbps for command. The S-band receivers can handle 10 Mbps from the payloads. This is sufficient for 3 landers (See

Appendix T for link budgets).

The MOM-E mission will conduct a technological demonstration of a reconfigurable micro-patch array with radio frequency micro-electromechanical systems (RF MEMS) switches. The design includes a number of small patch antennas that can be connected with very low noise switches. By flipping different switches, the effective length or area of the array changes, which alters the resonating frequency of the antenna. Ultimately, MOM-E could communicate on a number of frequencies while minimizing loss of performance, improving its modularity.

5.0 Systems

5.1 Mass Budget

The Proton-M rocket selected has the potential to carry 3,350 kg of payload into orbit. The mass budget was created based on this potential payload, and a 30% contingency was included.

Subsystem	Mass (kg)	30% Contingency (kg)	Total Mass (kg)
Payload	1153.85	345.16	1499.01
Structures	768.44	230.53	998.97
Power	79.97	23.99	103.96
Thermal	78.84	23.65	102.49
ADCS	389.79	116.88	506.67
C&DH	8.75	2.63	11.38
Communications	33.59	10.08	43.67
Total	**2513.23**	**752.92**	**3266.15**

Table 6: Mass Budget

5.2 Risk Analysis

Table 7 shows the main critical risks.

Major Failure Mode	Subsystem
Star Tracker Failure	ADCS
Communication Link Failure	Communications
Structure Failure During Launch	Structures
Excessive Radiation	Thermal/Radiation
Kickstage Failure	ADCS
Payload Separation Failure	Payload

Table 7: Critical Risk Matrix

A complete risk analysis table, along with mitigation methods, can be found in Appendix U.

5.3 System Timeline

The inaugural MOM-E will take approximately 9 years to develop, manufacture, test, and launch. After pre-mission procedures, MOM-E will be launched from the surface of the Earth. It will take about 6 days for it to reach its orbit, where, upon arrival, it will deploy its orbiter and lander payloads. MOM-E will then remain in its orbit for 3 years and serve as a communications relay to Earth for deployed payloads. Additionally, it will observe and record data using onboard sensors and instrumentation. At the end of 3 years, if all mission objectives and goals have been met, MOM-E's attitude determination and control system will be shut down and it will de-orbit.

R&D, Production and Testing: 6.5 Years	
Design & Development	4 years
Manufacturing	1.5 years
Integration and Testing	1 year
Preparation and Launch: 36 days	
Pre-Launch Preparation	30 days
Launch to LEO	1 day
Transfer Orbit to Moon	5 days
Lunar Orbit Destination: 3 years	
Deploy Payloads	<1 week upon arrival in Lunar orbit
Perform Communications Relay	3 years
Close Operations and De-Orbit	1 week
Total Mission Length	9.5 years

Table 8: Complete Timeline

6.0 Business Viability

The customer base chosen for the mission was derived from a market analysis that identified past and future space missions to the Moon. The potential customers include those building Lunar landers or orbiters similar to ones competing in the Google Lunar X-Prize (GLXP) competition. Several upcoming missions have been identified that use remote sensing instrument packages to study the Lunar surface; therefore, there is great interest in transporting scientific instruments to the Moon.

The Research, Development, Test and Evaluation (RDT&E) Work Breakdown Structure (WBS) provides a framework for non-recurring cost drivers of the MOM-E mission. The costs of RDT&E are broken into 4 different groups: program level costs, space segments, ground segments, and operation and maintenance. Production and Operation & Maintenance WBSs provide the framework for the recurring cost. Combined, the breakdowns organize the total mission costs and identify the cost drivers (Appendix V).

The Cost Estimating Relationship (CER) estimates the non-recurring and recurring costs of the proposed MOM-E mission. The non-recurring cost includes all of the RDT&E operations, while the recurring cost includes all of the costs involved in the completion of the theoretical first mission. The CER model uses the mass of each spacecraft subsystem as an input into a regression algorithm based off historical data from previous NASA missions. The CER estimates the non-recurring cost at $320 million and the recurring cost at $166 million.

To create a successful business model, the revenue from the first mission must equal the recurring cost in order to prevent losses. To ensure this, the recurring cost is fairly distributed among the customers based on an allocation model that uses both the mass and the height of the customers' payloads. The types of customers that are targeted include: small lander/orbiter, medium lander/orbiter, large lander/orbiter, instrument, and CubeSat. Each customer type is standardized with a specific mass and height derived from a heritage study.

Twenty population scenarios were studied to determine the average price for each type of customer. Each population scenario includes different combinations of customer types that fill the payload bay to capacity either in mass or height. The average price is quoted to provide the customer with an initial cost estimate, and is subject to change due to the dependence of the customer prices on the specific population.

Customer Type	Competitor Price (USD)	MOM-E Quoted Price (USD)
"1U" CubeSat (1 kg)	Currently None	$1.1M/"U"
Instrument (100 kg)	$25 M	$19 M
Small Orbiter (150 kg)	$25-35 M	$26 M
Medium Orbiter (350 kg)	$40-50 M	$45 M
Large Orbiter (500 kg)	$50-60 M	$62 M

Table 9: Cost Comparison

To make the mission profitable, the team created a business model that converts efficiency savings into profit. The efficiency savings are modeled using a learning curve which capitalizes on a 5% savings after every successive unit produced. Therefore, 70% of the efficiency savings are converted into a profit while 30% of it is used to reduce the customer's cost. A total accrued profit of $22 million is expected after 5 mission cycles.

For details on business models used in this analysis, please refer to Appendix V.

7.0 Conclusion

The students of Aerospace Engineering 483 designed MOM-E as a proof-of-concept mission for a new class of spacecraft in which a mothership is responsible for transporting customer payloads to orbit and serving as a communications relay for those payloads. Currently, MOM-E is scaled for Lunar exploration, targeting the growing robotics space exploration industry. Through design iteration and trade studies, the team has found that the Lunar-specific mission is conceptually feasible if sufficient funding is acquired.

References

http://blog.grandtrunk.net/2004/07/practical-compressor-test/?network=1000&decomp=1#ranking

http://cassini-huygens.jpl.nasa.gov/cassini/Spacecraft/command.shtml

http://h10144.www1.hp.com/case-studies/International_Space_Station.htm?jumpid=re_R2612_us/en/any/Corp/do-amazing-us:explore:networkingEADS

http://www.astronautix.com/lvs/proton.htm

http://www.astronautix.com/stages/blockd.htm

http://www.b14643.de/Spacerockets_1/East_Errope_2/Proton-M/Description/Frame.htm http://www.baesystems.com/ProductsServices/bae_prod_s2_rad6000.html

http://www.broadreachengineering.com/mirideon.html

http://www.dcs.gla.ac.uk/~ross/Ethernet/protocol.htm

http://www.friends-partners.org/partners/mwade/lvs/pro2kdm3.htm

http://www.futron.com/pdf/resource_center/white_papers/FutronLaunchCostWP.pdf

http://www.govcomm.harris.com/solutions/products/000112.asp

http://www.ilslaunch.com/protonmpg/

http://www.imation.com/en-us/

http://www.marsrover.nasa.gov

http://www.maxwell.com/microelectronics/products/_sbc/index.html

http://www.precidip.com/catalog/product.asp?c=102&p=351&i=1710&q=802-10-NNN-30-507101

http://www.precidip.com/catalog/product.asp?c=11&p=246&i=795&q=813-SS-NNN-30-XXX101

http://www.spacemicro.com/space_div/se_div.htm#proton400kl

http://www.thermacore.com/products/advanced-solid-conduction

Appendix A: Mission Requirements

- FNC-SYS-001: The spacecraft shall support a Google Lunar X-Prize mission
- FNC-SYS-002: The spacecraft shall communicate with Earth
- FNC-SYS-003: The spacecraft shall take health data checks throughout mission lifetime
- FNC-SYS-004: The spacecraft shall release and/or activate payloads upon reaching Lunar orbit
- FNC-SYS-005: The spacecraft shall release and/or activate payloads upon reaching circular Lunar orbit
- OPR-SYS-001: The spacecraft shall comply with all applicable FAA and FCC regulations
- OPR-SYS-002: The spacecraft shall support full operations for a minimum of 3 years
- OPR-SYS-003: The spacecraft shall support deployable and non-deployable payloads
- OPR-SYS-004: The spacecraft shall withstand environmental exposure throughout mission lifetime
- BUS-001: The team shall create an affordable transportation system for private companies, international governments, and coalitions of universities to explore space

- BUS-002: The team shall adhere to all international regulations pertaining to the spacecraft design, testing, launch, and operations
- BUS-003: The team shall design a spacecraft with modular payload bays
- PLD-001: Insure that MOM-E shall be capable of supporting small, medium, and large payloads
- PLD-002: A standard interface shall be created in order to promote modularity
- PROP-001: The spacecraft shall reach low Lunar orbit
- CDH-001: C&DH shall be capable of taking and processing health data of the payloads and mothership during transit
- CDH-002: C&DH shall be capable of supporting sensor package payloads through a standardized interface and protocol
- CDH-003: C&DH shall be capable of support communication relay operations
- CDH-004: C&DH shall be capable of taking health data from the mothership during mission operations
- CDH-005: C&DH shall store received payload data until it can be relayed back to Earth
- CDH-006: C&DH shall interface with all GNC sensors and support their corresponding protocols
- COMM-001: The spacecraft shall be capable of transmitting data collected from 3 landers to Earth
- COMM-002: The spacecraft shall be capable of receiving telemetry data from 3 landers
- COMM-003: The spacecraft shall be capable of receiving command data from Earth and relay it to 3 landers
- GNC-001: GNC shall maintain the desired orbit altitude throughout the mission lifetime
- GNC-002: GNC shall maintain orientation and pointing for proper communications with Earth and the payloads when necessary
- GNC-003: GNC shall orient the spacecraft to allow for proper deployment of payloads
- THM-001: Thermal shall keep mothership control area within survivable range for all components
- THM-002: Thermal shall keep all components within operational temperature range during their operation
- THM-003: Thermal shall limit Total Ionizing Dose to a survivable amount for all components
- THM-004: Thermal shall limit Acute Dose to a survivable amount for all components
- STR-001: The structure shall withstand loads of 5 g's with a safety factor of 1.25
- STR-002: The structure shall house all subsystems
- STR-003: The structure shall fit inside the fairings of the launch vehicle
- STR-004: The structure should separate from the kickstage
- EPS-001: EPS shall be scalable for future missions
- EPS-002: EPS shall provide power to mothership for a minimum of 3 years
- EPS-003: EPS should provide continuous power during normal operations
- EPS-004: EPS shall regulate power to all subsystems
- EPS-005: EPS shall power deployment of kickstage
- EPS-006: EPS shall provide a modular power system to payloads for deployment and power in transit

- EPS-007: EPS shall be operational in eclipse and in the sun

Appendix B: Payload Heritage Studies

One payload type that was analyzed for potential MOM-E payloads was orbiters. Orbiters have a large range of potential uses and are relatively simple in design and operation. Most orbiters would be able to fit into a MOM-E payload bay and with their extensive flight heritage they prove to be a highly plausible option for potential customers. Many different orbiters were considered for the MOM-E heritage study, including the option of launching many CubeSats out of an NPSCul-Lite for their affordability and low mass, volume, and data rate requirements. Data of those orbiters can be seen below.

Satellite	Mass(kg)	Volume(m^3)	Data Rate(kbps)	Type
Chandrayaan-1	1380	3.375	50,000	Orbiter/Probe
LRO	1916	15.444	5000	Orbiter
LCROSS	891	10.613	1000	Orbiter/Impactor
Chang'e 1	130	7.48	36	Orbiter
Kaguya	2914	18.522	10,000	Orbiter w/ 2 Sats
Smart 1	367	1	500	Orbiter
Cassini/Huygens	2150	108.8		Orbiter/Probe
CubeSat	5	0.01	10	Orbiter
30U NPSCul	77	0.07	100	Obiters/Launcher

Appendix Table 1: Orbiter Heritage

Another payload type that was analyzed for potential MOM-E payloads was rovers/landers. They have the greatest ability to analyze a celestial body's features. They are more complicated, costly, and failure prone than orbiters and are limited to certain areas of the surface. However, they have a much higher potential for major scientific breakthroughs than orbiters. This makes rovers/landers a very likely large payload on a MOM-E mission. Also, with the inception of the Google Lunar X Prize, the demand for Lunar rovers is elevated. A few different rovers and landers were considered for the MOM-E trade studies. Data of those orbiters can be seen below.

Name	Mass(kg)	Volume(m^3)	Data Rate(kbps)	Type
Spirit/Opportunity	185	3.10	200	Rover
Spirit/Opportunity	685	8.82	0.5	Lander
Phoenix	350	3.89		Lander
Pathfinder	816	0.89	1	Lander
Pathfinder	10.5	0.09	0.125	Rover
Team Frednet	150	0.57	9	Lander+Rover

Appendix Table 2: Rover/Lander Heritage

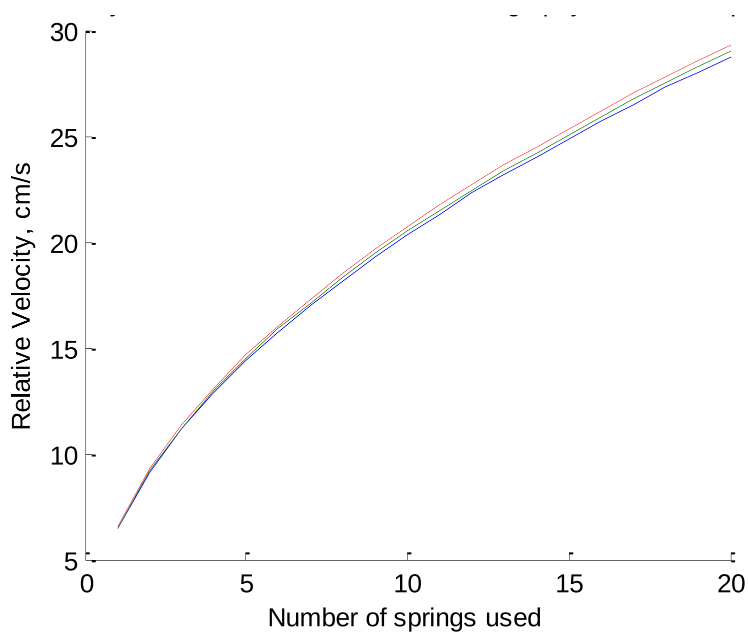
Appendix C: Lightband Ejection Force

The equation that relates the mass, relative velocity, and number of springs is provided by the Mark II Lightband manufacturer (Planetary Systems Corporation) as

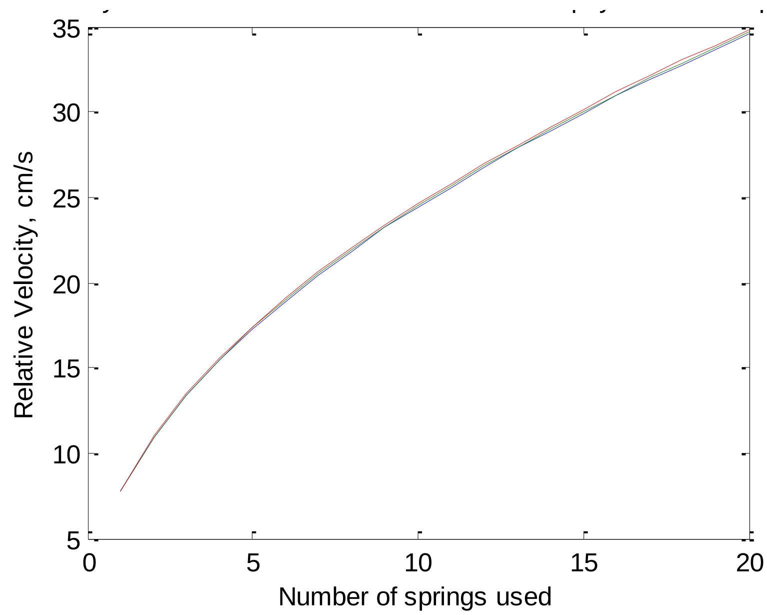
$$S = \frac{m * M}{m + M} * \frac{v^2}{2 * n * E}$$

Where *S* is the number of separation springs used, *m* is payload mass, *M* is final stage mass, *v* is the relative velocity between the payload and MOM-E, *n* is the efficiency (.9 ± .03), and *E* is the stored potential energy of a separation spring, provided by Planetary Systems Corporation (PSC) testing.

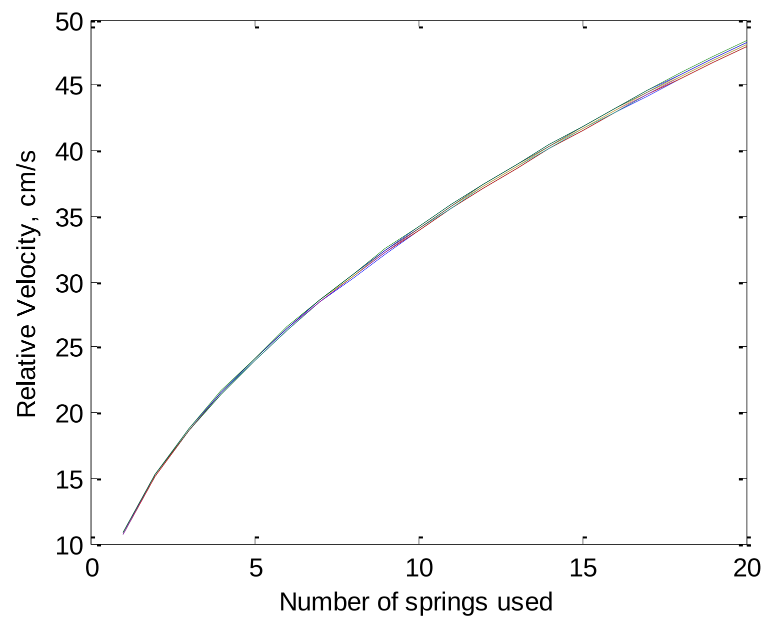
As payloads get ejected, the final stage mass, M, in the above equation will change. This will cause the difference in relative velocity between the first and last payload ejection to be only millimeters/s, which is much slower than the relative velocity of the payload and MOM-E. Due to this small change, the team will not further analyze all the scenarios, which would be necessary to complete before launch and payload ejection.



Appendix Figure 1: Large Payload Ejection

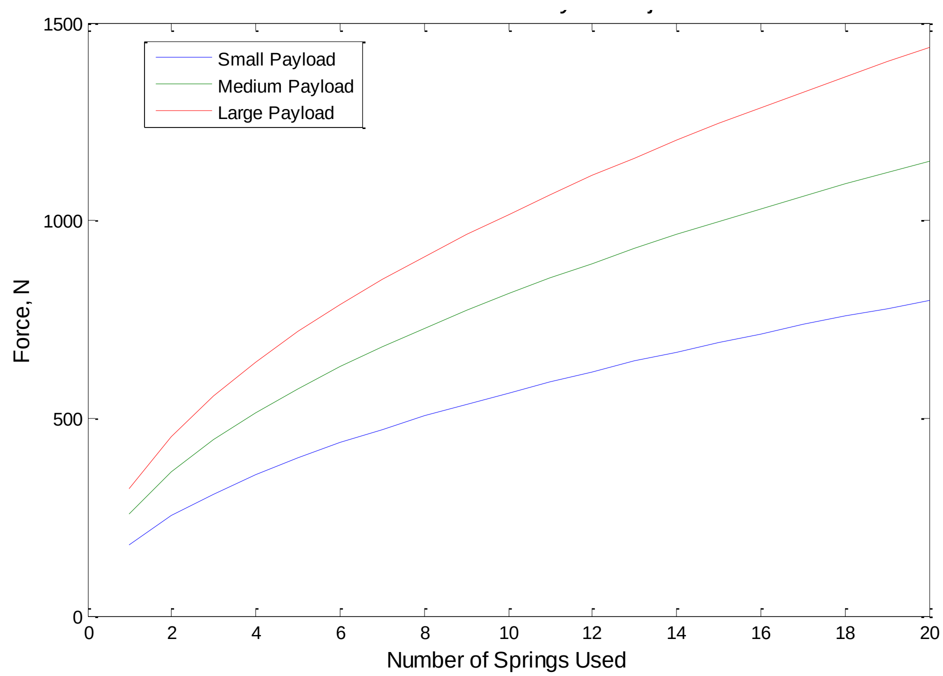


Appendix Figure 2: Medium Payload Ejection



Appendix Figure 3: Small Payload Ejection

Now that the relative velocity is known, the impulse caused to MOM-E can be calculated. Planetary Systems Corporation has provided test data for the time it takes to eject, and with this data the team can calculate the force. If the center of mass of the payload is not in line with the center of the ejection system, there will be a rotational element with the ejection. However, this will not be analyzed in this report due to the varying nature of the payloads. To calculate the force applied to MOM-E, this analysis assumes that the force is applied directly normal to the attachment point. Appendix Figure 4 shows the results of this analysis.



Appendix Figure 4: Force Caused by Payload Ejections

Appendix D: Mass-Propellant Relation

For an analysis, the team would like to find how much propellant is needed to land an object on the Moon. To find the absolute minimum change in velocity (ΔV) needed to land a craft on the Moon, the team assumed that the initial Lunar orbit of 300km is circular, that there is a Hohmann transfer orbit to the altitude of the Lunar surface, and that the Moon is a perfect sphere. More complicated maneuvers were assessed, but the results in this case differed by less than 7 percent. Applying that

$$v_{circular\ orbit} = \sqrt{\frac{G * M}{r}}$$

where *G* is the gravitational constant, *M* is the mass of the Moon, and *r* is the distance from the orbit to the center of the Moon. The required ΔV from to Lunar orbit to being stationary on the Lunar surface is -1809.6 m/s. Using the rocket equation

$$m_{total} = m_{dry} * e^{-\left(\frac{\Delta V}{V_0}\right)}$$

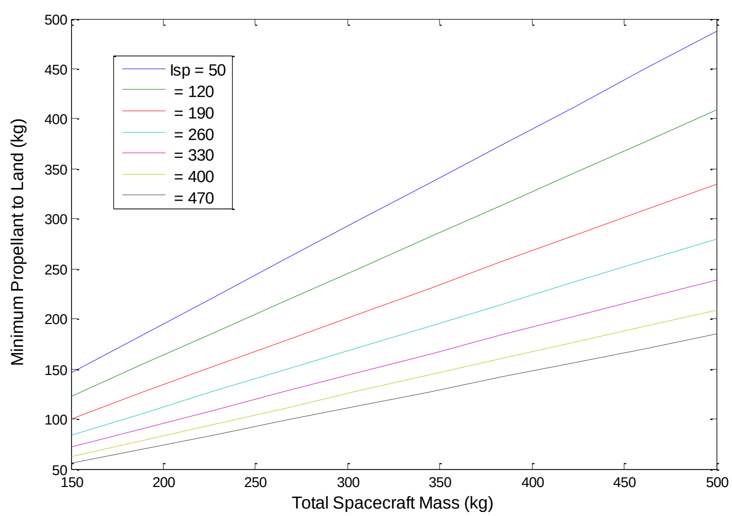
where

$$m_{total} = m_{dry} + m_{propellant},$$

and

$$V_0 = I_{sp} * g_{Earth}.$$

Arranging this equation gives us the mass of propellant needed when compared to the total payload mass and specific impulse (I_{sp}) of the payload thrusters. Team Frednet, a team going for the Lunar-X prize, has developed a lander which they claim to have a specific impulse of 320 seconds. A comparison of propellant mass to overall mass with several different specific impulses is shown in Appendix Figure 5. Using this as a baseline, one may see that a large, 500 kg payload is going to need to use about half its mass as just propellant to slow down and land. This still leaves over 200 kg that can be used to conduct scientific research.



Appendix Figure 5: Minimum Propellant Mass as a Function of Total Payload Mass

Appendix E: MOM-E Payload User's Manual

Thank you for your interest in this venture. This manual details the constraints and services your payload will be given if you choose to fly with us on our MOM-E mission.

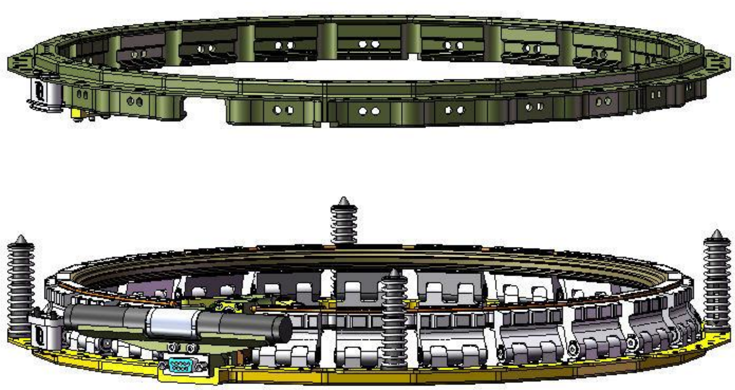
NOTE: To avoid costly design alterations, customers should completely understand this document before purchasing space on the MOM-E mission.

MOM-E Payloads Defined

1. One primary payload
   500kg and (1.4 m height x 1.8 m diameter) constraints
2. Two secondary payloads
   500kg and (1.4 m height x 1.8 m diameter) constraints
3. X number of tertiary payloads
   Mass and volume constraints variable

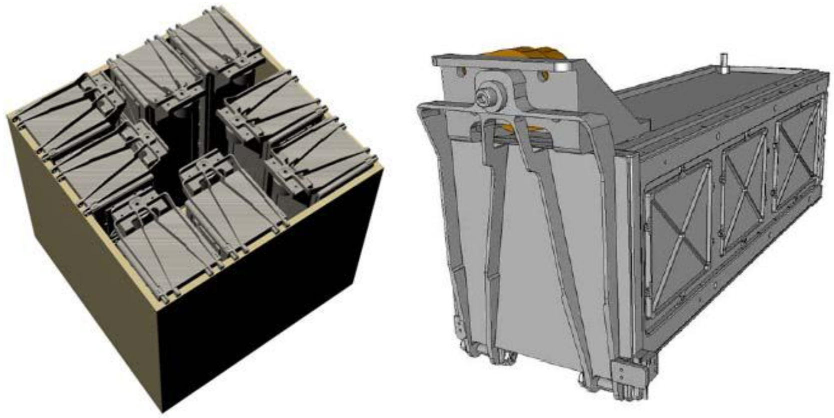
Services Provided

1. This venture will provide the payload with a ride on the mothership to the Moon.
2. The primary payload can negotiate the initial Lunar orbit inclination. Possible inclinations are from -8 to 28^{o}.
3. All payloads can specify where on the orbit they would like to be launched (if applicable).
4. A Mark II Motorized Lightband will be provided to all customers that require deployment. Customers are guaranteed a separation signal, the power to separate, and a customized Lightband electrical connector for health checks.



Appendix Figure 6: Mark II Motorized Lightband

   Additional details on the Mark II Lightband can be found at
   http://www.planetarysystemscorp.com/download/2000785B_UserManual.pdf
5. CubeSat customers are guaranteed a ride in a Poly-PicoSatellite Orbital Deployer (P-POD). The P-POD will be housed within a Naval Postgraduate School CubeSat Launcher (NPSCuL-Lite). Deployment from the launcher will occur while launcher is attached to the mothership.



Appendix Figure 7: NPSCuL-Lite and P-POD Launcher

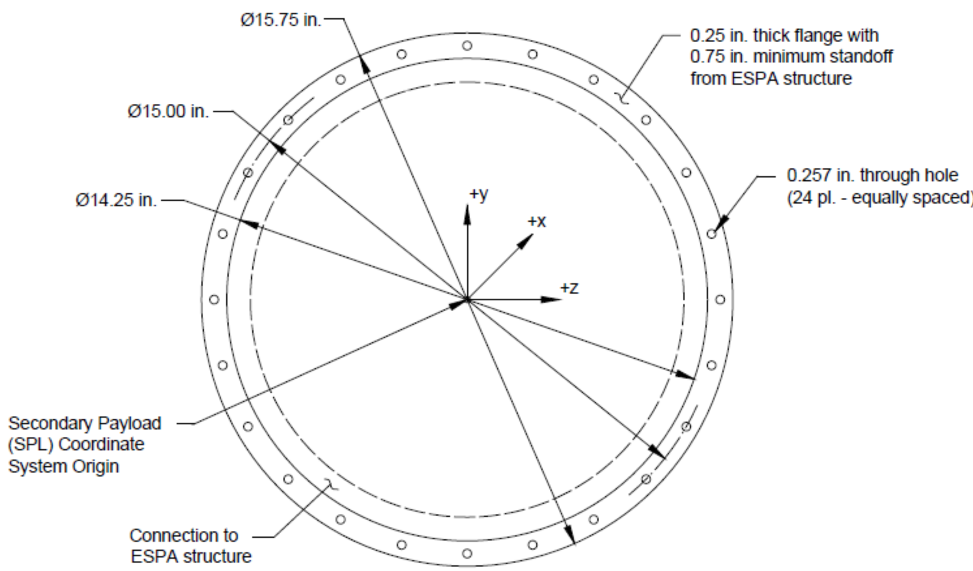
More information on the NPSCuL can be found at
http://www.dtic.mil/cgi-bin/GetTRDoc?AD=ADA501503&Location=U2&doc=GetTRDoc.pdf
More information on the standard CubeSat and P-POD configuration can be found at
http://cubesat.org/images/developers/cds_rev12.pdf

6. Instrument payloads can be guaranteed pointing to within 1^{o}. Pointing orientations should be coordinated with this venture

Payload Constraints

1. Acceptance of all payloads is subject to approval by this venture
2. The secondary and tertiary payloads shall not affect the primary payload's mission orbit requirements.
3. This venture will manifest the payloads and ensure they meet the mission integration schedule. The secondary and tertiary payloads shall meet all mission schedules and shall not impact the primary payload in any negative manner. That is, the primary payload drives the launch schedule and the secondary payloads must meet this schedule. The secondary and tertiary payloads should be ready to enter the mission specific (typically 36 month) integration schedule per the mission unique launch vehicle schedule. This venture reserves the right to cancel missions upon failures to obtain enough payloads for launch.
4. Manifesting of tertiary payloads will be considered only for missions that have excess volume and mass margin. This venture may withdraw tertiary payloads if any of these margins are unexpectedly reduced.
5. All payloads shall not exceed the volume allotted. This volume includes the Mark II Motorized Lightband if deploying. Payload volumes exceeding their volume constraints will be negotiated with this venture
6. All CubeSat payloads shall conform to the 3u CubeSat standards set by California Polytechnic State University.
7. All payloads shall not exceed the mass allotted. This mass includes the Mark II Motorized Lightband if deploying. Payload volumes exceeding their mass constraints will be negotiated with this venture

8. The primary and secondary payloads shall provide a volume and mass estimate to within 10% a minimum of 12 months prior to launch so that tertiary payloads can be determined. Payloads that fall outside their estimated range may not be permitted to launch.
9. All payloads not planning to deploy shall be capable of mounting on an ESPA connection. The ESPA connection is detailed below.



Appendix Figure 8: ESPA Connection

10. All payloads shall not present any hazard to the mothership or other payloads in the form of EMI radiation, contamination, ordnance, etc. Mission specific exceptions may be coordinated.
11. Tertiary payloads may be given a mass target upon acceptance (determined by this venture) that must be met. Major deviations from the pre-determined mass target may not be acceptable.
12. There shall be no standard access to payloads after encapsulation.
13. Payloads are only guaranteed one customized Lightband electrical connector. Additional connectors or other electrical interfaces, including umbilical lines, will be negotiated through this venture
14. Payloads shall not require telemetry or commands during launch.

Appendix F: Structural Mass Budget

Appendix Table 3 represents the structural mass budget for the MOM-E mission. The last column represents the total mass of each structural component with the standard preliminary design contingency of 30%. Refer to Appendix I for each component's location on the mothership and design.

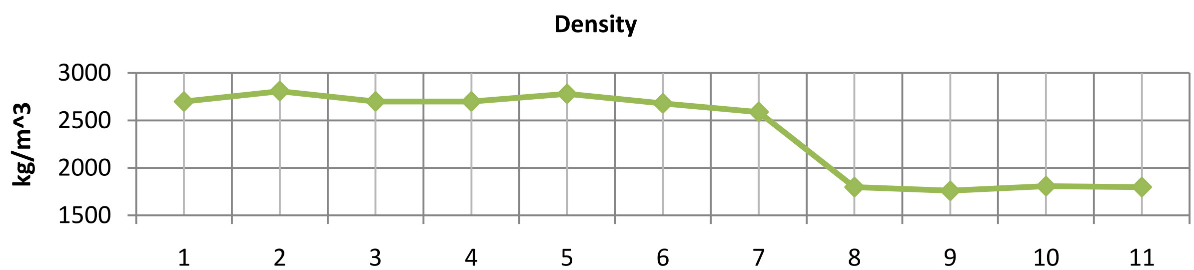
Structural Component	Mass (kg)	Quantity	Total Mass (kg)	Total Mass with Contingency (kg)
Bottom Module	59.92	1	59.92	77.90
Middle plate	32.25	1	32.25	41.93
Top Module	26.52	1	26.52	34.48
Mothership Interface	40.96	1	40.96	53.25
Payload Plate	74.77	3	224.31	291.60
Truss	23.8	9	214.2	278.46
Rings	6.76	3	20.28	26.36
Kickstage Interface	100	1	100	130.00
Connections and Fasteners	50	1	50	65.00
Total			**768.44**	**998.97**

Appendix Table 3: MOM-E Structural Mass Budget

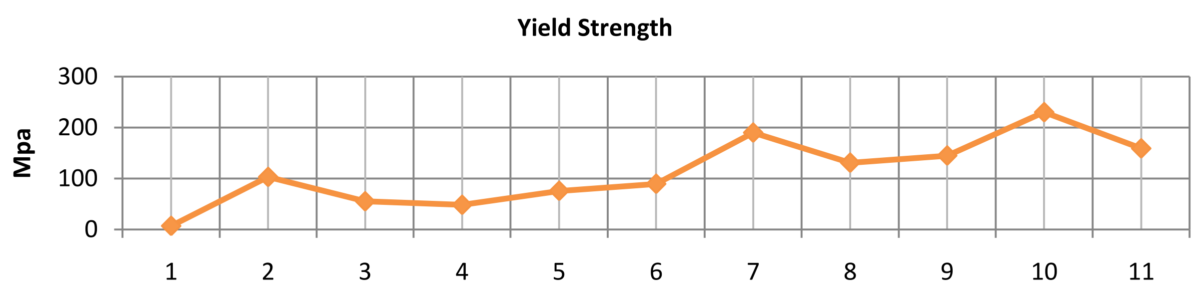
Appendix G: Structural Materials

Number	Material
1	Aluminum
2	7075 Aluminum
3	6061 Aluminum
4	6063 Aluminum
5	2024 Aluminum
6	5052 Aluminum
7	2090 Aluminum
8	AM60 Magnesium
9	AZ10A Magnesium
10	AZ80A Magnesium
11	AZ91D Magnesium

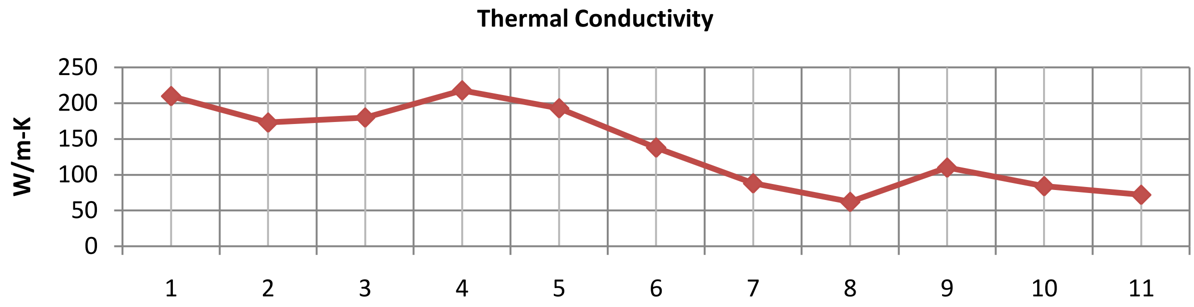
Appendix Table 4: Considered Materials



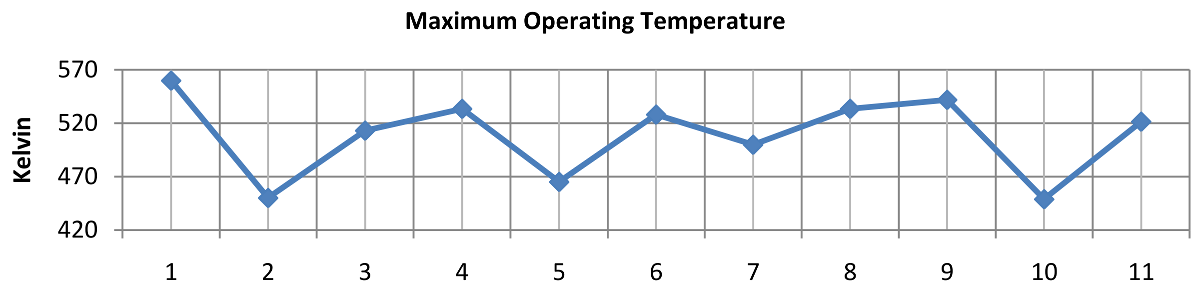
Appendix Figure 9: Density Plot



Appendix Figure 10: Yield Strength Plot



Appendix Figure 11: Thermal Conductivity Plot



Appendix Figure 12: Maximum Operating Temperature Plot

Cost of Aluminum 7075 and Titanium Calculations

$$m_{material} = \rho_{material} v_{material}$$

Where:

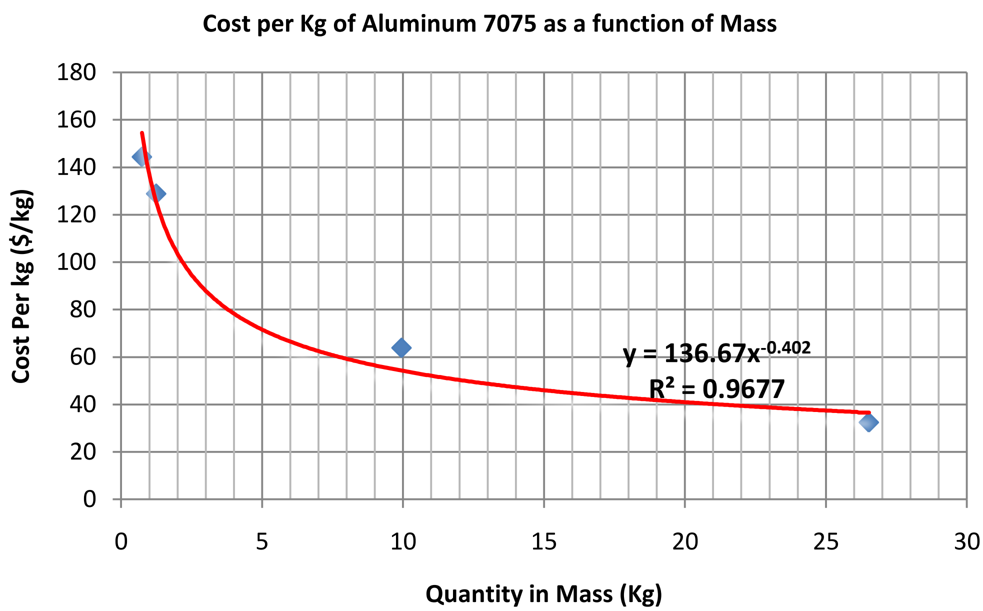
$$\rho_{al} = 2.81\ \frac{g}{cc}$$

$$\rho_{Ti} = 4.50\ \frac{g}{cc}$$

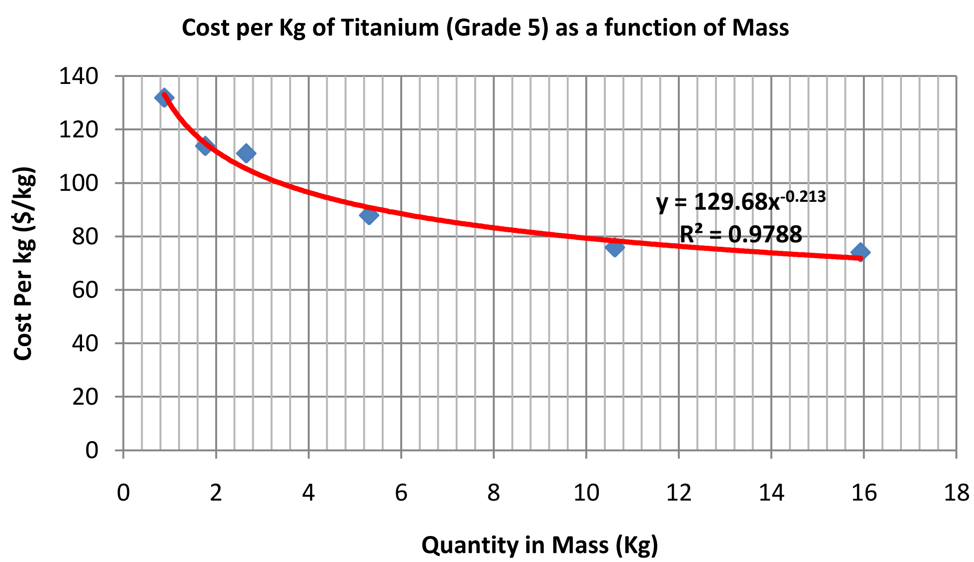
From McMaster Carr, the team can find a range of prices for sheets of aluminum and titanium alloys. Calculating the price for a given volume of a sheet allows us to calculate its given mass assuming the team knows the density for material. Dividing the price that McMaster is charging for that given sheet by the calculated mass will yield the Cost per kilogram of Material.

$$C = \frac{P}{m_{material}}$$

where C is in dollars per kilogram of material, and P is price of the sheet of the aluminum or titanium.

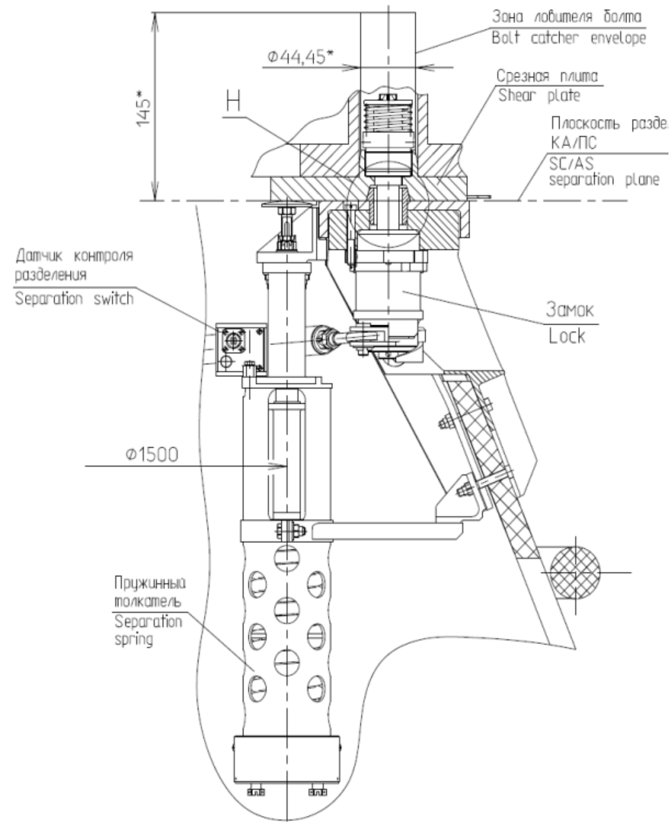


Appendix Figure 13: Cost per kg of Aluminum



Appendix Figure 14: Cost per kg of Titanium

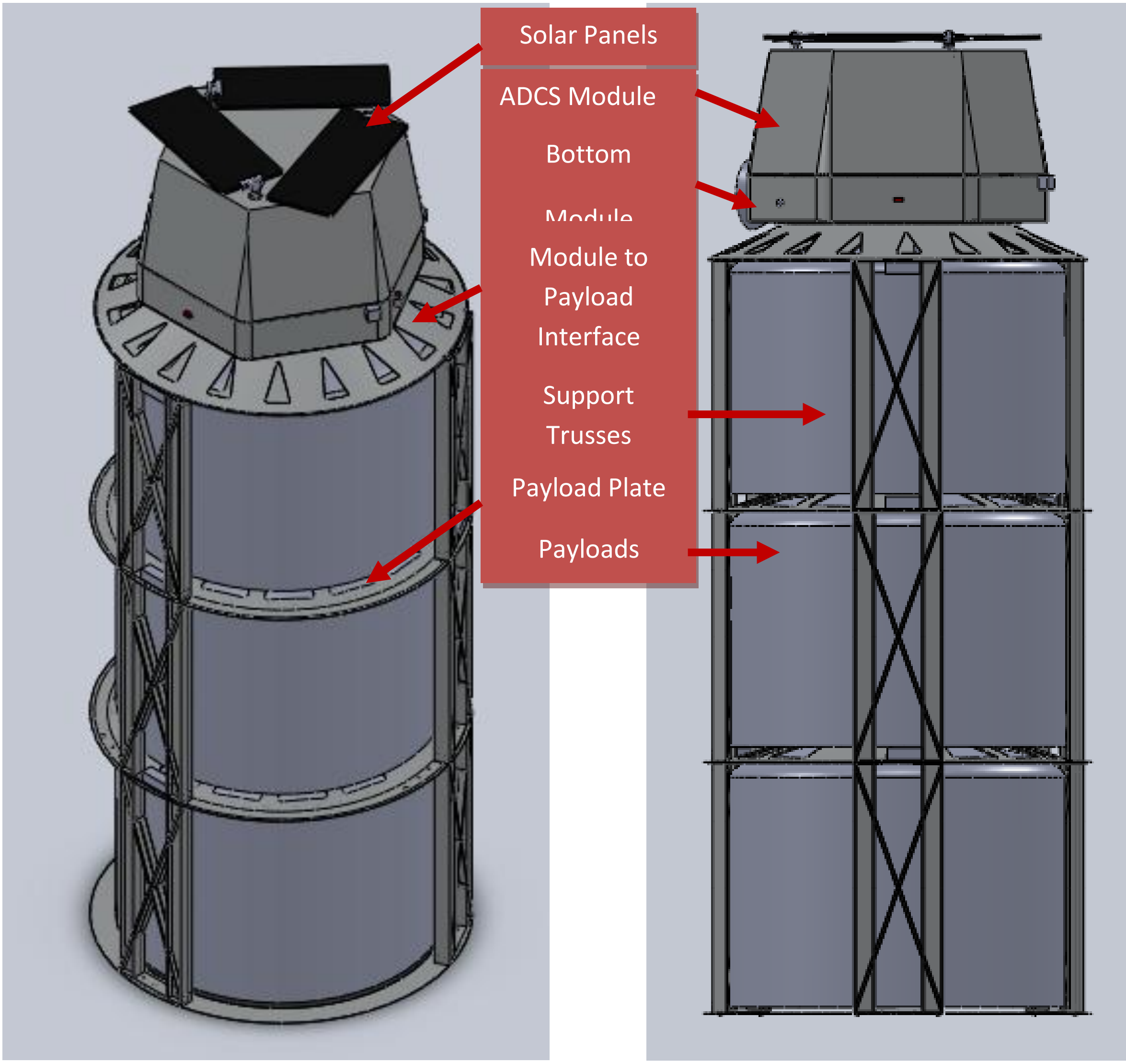
Appendix H: Lock-and-Spring Mechanism



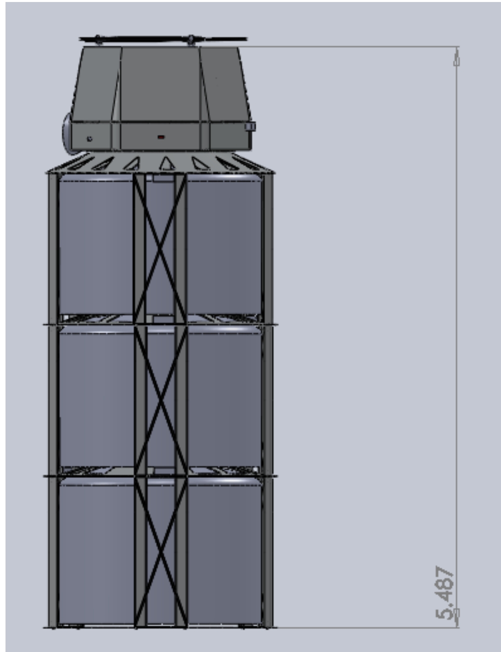
Appendix Figure 15: Lock-and-Spring Drawing

Each adapter system utilizes interface rings, 3 latches and push springs, and umbilical connectors. An electrical signal is sent to the adapter system for a particular interface section through the umbilical connects. When this is done the 3 locks and push springs simultaneously release to separate the intended section from the mothership. When this occurs, the umbilical connector disconnects the electrical relay for that section, and the process is repeated for the other interfaces.

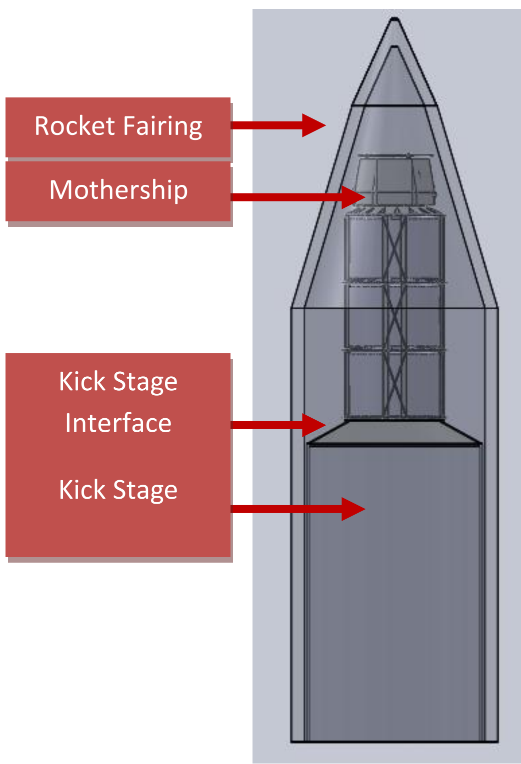
Appendix I: Mothership Layout



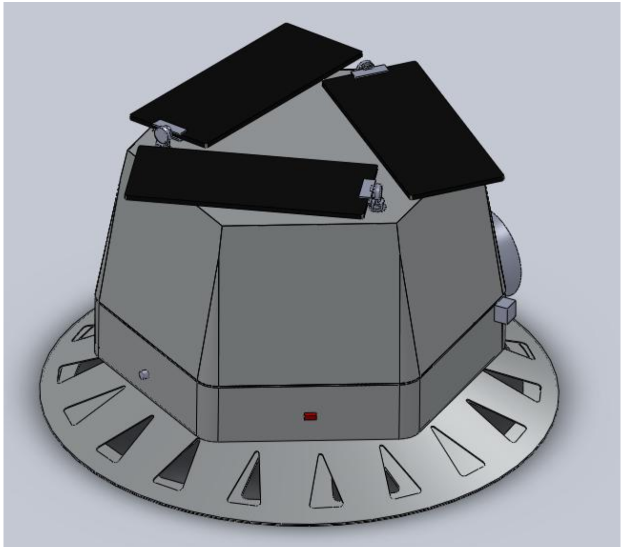
Appendix Figure 16: Mothership at Launch

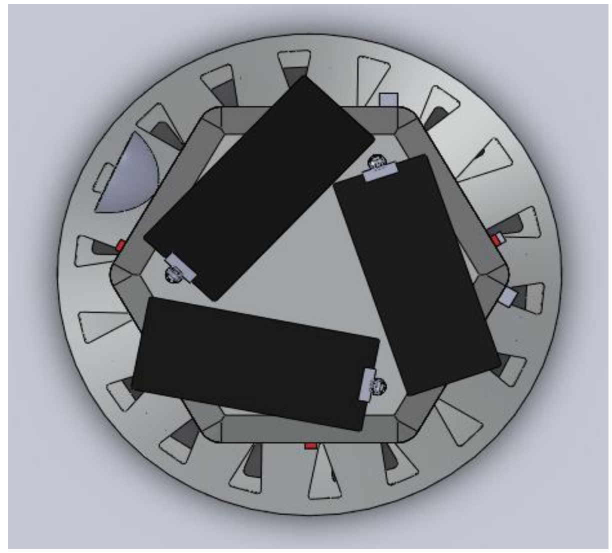


Appendix Figure 17: Height

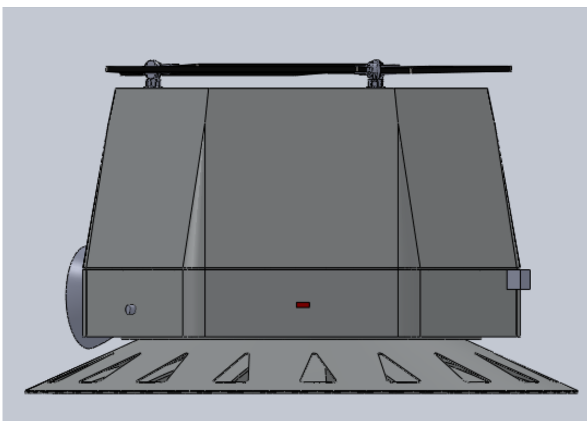


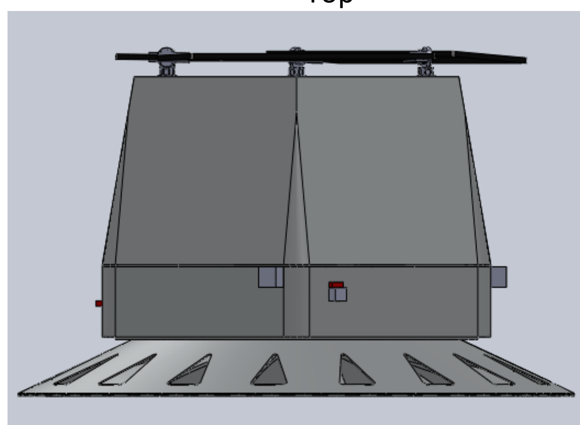
Appendix Figure 18: Inside Fairing





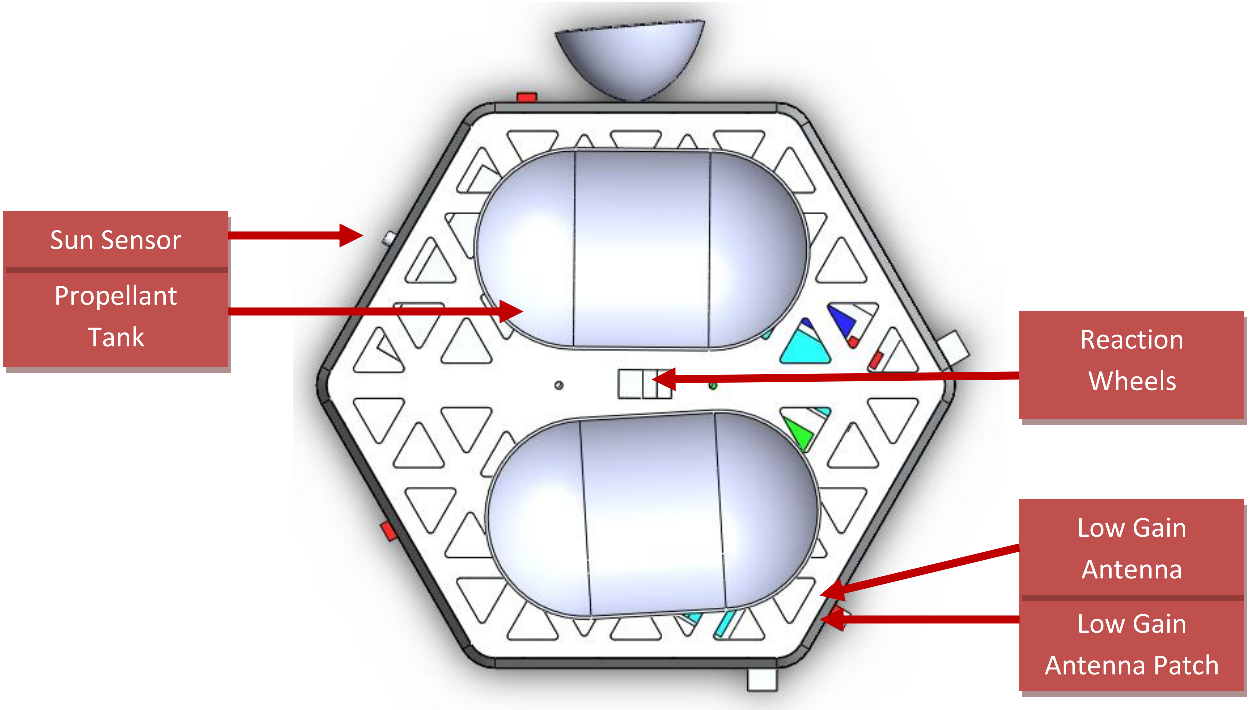
Isometric | Top



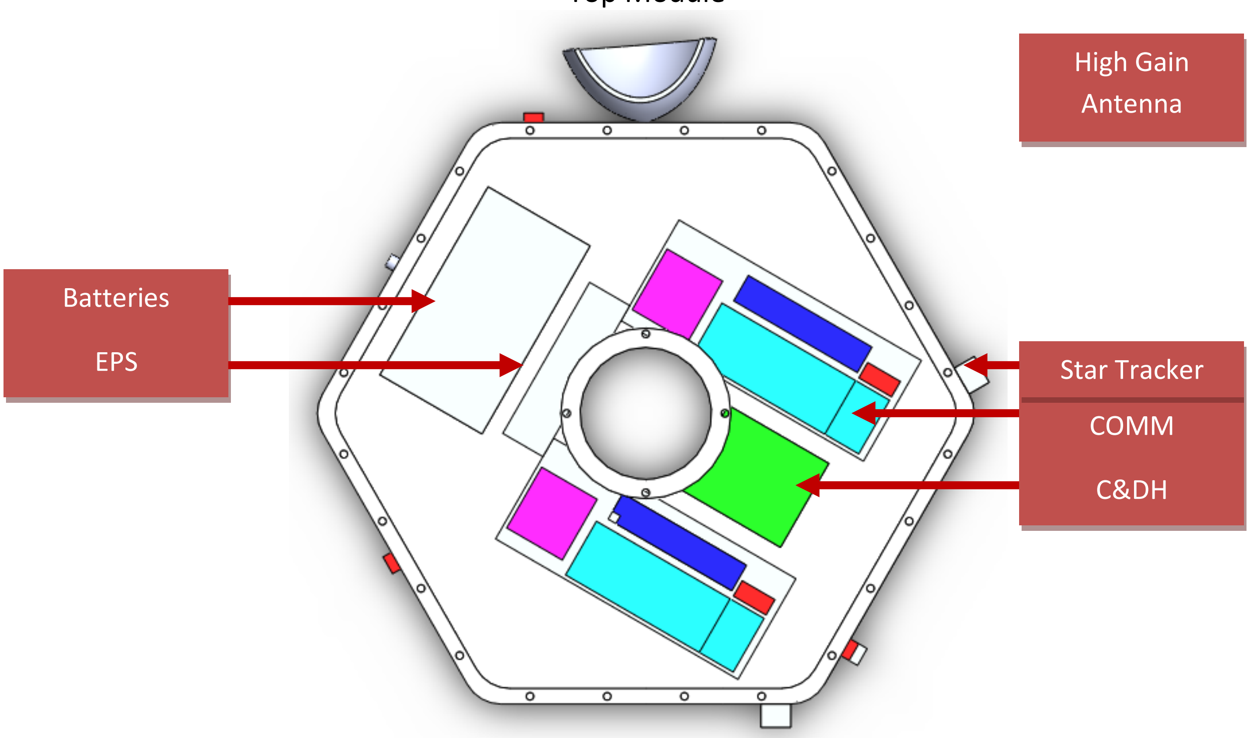


Front | Right

Appendix Figure 19: Mothership After Payloads Deploy



Top Module



Bottom Module

Appendix Figure 20: Internal Layout

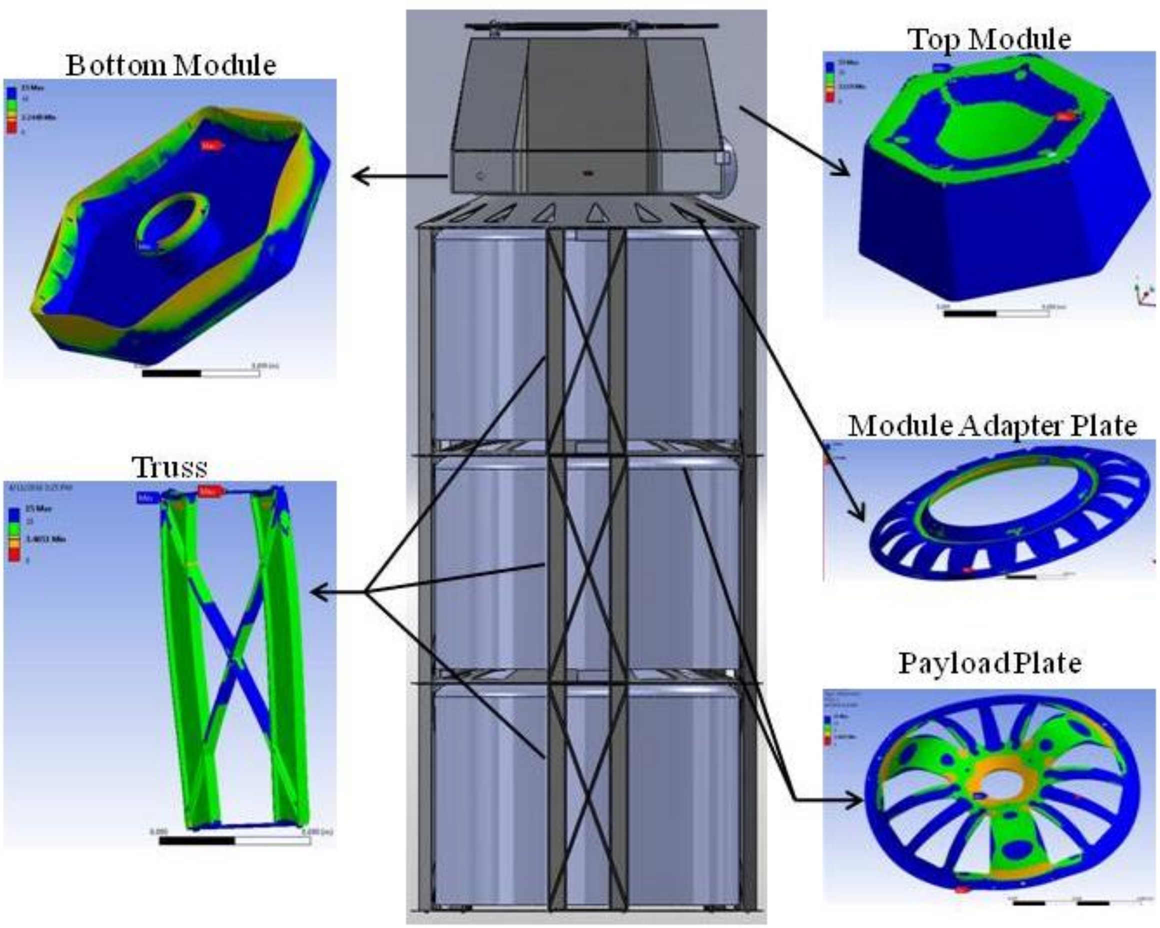
Appendix J: Stress Analysis Breakdown

Stress analysis was done on the components of the mothership to ensure no components would fail. Analysis was done using ANSYS and the minimum requirement was a safety factor of 1.25. The table below lists the safety factors that ANSYS calculated.

Component	Load Applied (kN)	Safety Factor
Top Module	2.25	3.119
Bottom Module	52	2.2448
Module Adapter Plate	55	2.1580
Truss	150	3.4651
Payload Plate	25	1.3847

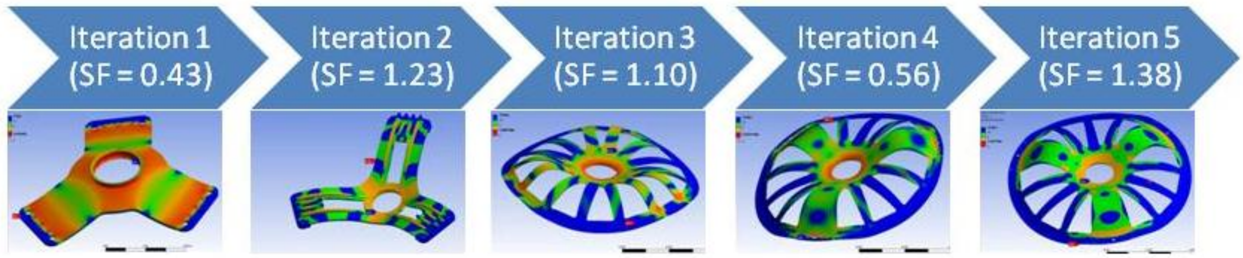
Appendix Table 5: Safety Factors

It should be noted that the safety factors for the truss and payload plate were calculated with the worst case loads since there are multiple instances of these components in the mothership. All the components satisfy and exceed the minimum requirement safety factor of 1.25.



Appendix Figure 21: Major Components of Mothership

All components except payload plate immediately satisfied the minimum safety factor requirement. The payload plate went through 5 different design iterations before it satisfied the minimum safety factor requirement. Iterations one through 3 were drafted when the payload was still mounted above the mothership and therefore would sustain a high compression stress. Iterations 4 and 5 were drafted after the payload was moved to underneath the mothership. The last iteration successfully satisfied the minimum safety factor requirement of 1.25.



Appendix Figure 22: Iterations of Payload Plate

Appendix K: Power Budgets

Power Budget	Earth Tx Comm Power Consumption	Payload Rx Power Consumption	Tx and Rx from Earth Power Consumption
	Total Average w/ Cont (W)	Total Average w/ Cont (W)	Total Average w/ Cont (W)
Payload	0	0	0
Communications	44	38	82
Thermal	25	25	25
Power Systems	21	21	21
C&DH	49	49	50
ADCS	29	30	30
Mom.E TOTAL	**169**	**163**	**208**
Battery Power Used In Eclipse	**169**	**163**	**208**
Battery Power Used In Sunlight	**-183**	**-189**	**-123**

Appendix Table 6: Normal Modes of Operation Power Budget 1

Power Budget	Rx from Earth and Tx to Single Payload Power Consumption	Hibernation Power Consumption	Attitude and Stabilization Power Consumption
	Total Average w/ Cont (W)	Total Average w/ Cont (W)	Total Average w/ Cont (W)
Payload	0	0	0
Communications	37	20	0
Thermal	25	25	25
Power Systems	21	21	21
C&DH	49	20	32
ADCS	30	8	83
Mom.E TOTAL	**162**	**95**	**161**
Battery Power Used In Eclipse	**162**	**95**	**161**
Battery Power Used In Sunlight	**-190**	**-257**	**-191**

Appendix Table 7: Normal Modes of Operation Power Budget 2

Power Budget	Payload Deployment Power Consumption	Post-Payload Deployment Stabilization Power Consumption	Earth to Moon Transit Power Consumption
	Total Average w/ Cont (W)	Total Average w/ Cont (W)	Total Average w/ Cont (W)
Payload	0	0	0
Communications	44	44	20
Thermal	25	25	25
Power Systems	21	21	21
C&DH	49	49	20
ADCS	83	83	159
Mom.E TOTAL	**222**	**222**	**246**
Battery Power Used In Eclipse	**222**	**222**	**246**
Battery Power Used In Sunlight	**-130**	**-130**	**-103**

Appendix Table 8: Secondary Modes of Operation Power Budget

Solar Power Available (Watts)	413
Solar Cell Area(meters^2)	0.787
Solar Cell Efficiency	28.0%
Solar Intensity(Watts/meter^2)	1360
Inherent Degradation	0.975 (2.75% per year)
Battery Power Available (Watts)	450
Battery Efficiency	20%
Solar Array Efficiency	20%

Appendix Table 9: Input Parameters

Appendix L: Energy Budgets

Energy Budget	Earth Tx Comm	Payload Rx	Tx and Rx from Earth
	Energy Consumed (W*hr)	Energy Consumed (W*hr)	Energy Consumed (W*hr)
Payload	0	0	0
Communications	42	30	29
Thermal	2	2	2
Power Systems	47	47	47
C&DH	87	83	77
ADCS	36	31	23
Total Energy Consumed (Eclipse)	**213**	**192**	**177**
Energy Consumed/Energy Gained (Daylight)	**0.35**	**0.32**	**0.29**
Depth Of Discharge of Battery (using only battery)**	**0.12**	**0.11**	**0.10**
Amount of Energy Gained in one Sun Pass	**0.20**	**0.21**	**0.22**

Appendix Table 10: Primary Modes of Operation Energy Budget 1

	Rx from Earth and Tx to Single Payload	Hibernation	Attitude and Stabilization
	Energy Consumed (W*hr)	Energy Consumed (W*hr)	Energy Consumed (W*hr)
Payload	0	0	0
Communications	7	45	0
Thermal	2	2	2
Power Systems	47	47	47
C&DH	74	0	7
ADCS	18	15	31
Total Energy Consumed (Eclipse)	**147**	**70**	**87**
Energy Consumed/Energy Gained (Daylight)	**0.24**	**0.11**	**0.14**
Depth Of Discharge of Battery (using only battery)**	**0.08**	**0.04**	**0.05**
Amount of Energy Gained in one Sun Pass	**0.23**	**0.27**	**0.26**

Appendix Table 11: Primary Modes of Operation Energy Budget 2

Energy Budget	Payload Deployment Energy Consumed (W*hr)	Post-Payload Deployment Stabilization Energy Consumed (W*hr)	Earth to Moon Transit Energy Consumed (W*hr)
Payload	0	0	0
Communications	0	0	45
Thermal	39	55	0
Power Systems	47	47	47
C&DH	0	0	0
Guidance, Navigation, and Control	0	0	48
Total Energy Consumed (Eclipse)	86	102	140
Energy Consumed/Energy Gained (In Daylight)	0.14	0.17	0.23
Depth Of Discharge of Battery (using only battery)**	0.05	0.06	0.08
Amount of Energy Gained in one Sun Pass	0.26	0.25	0.23

Appendix Table 12: Secondary Modes of Operation Energy Budget

Battery	
Power Production (W)	450
Energy Available (W*hr)**	1800
Percent of Orbit in Sun	67

Appendix Table 13: Input Parameters

Appendix M: Thermal Calculations

Heat flux balance equations:

$$q_{EarthIR} = I_{EIR}F_{EIR}$$
$$q_{albedo} = \alpha\rho_{albedo}I_{solar}F_{albedo}$$
$$Q_{int} + q_{ext}A_{heat} = \varepsilon\sigma A_{surf}T_{surf}^4$$
$$q_{ext} = q_{albedo} + q_{EarthIR} + q_{solar} + q_{backload}$$

Radiation sources:

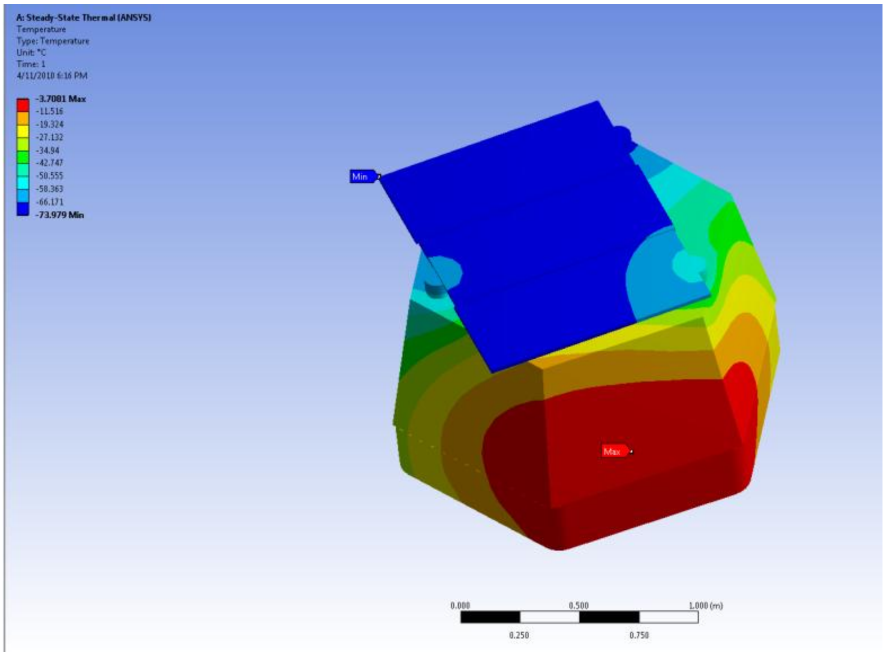
Sun => 1367 W/m^2 (average at 1AU)
Earth => 231 W/m^2
Moon => 430 W/m^2

Matlab code:

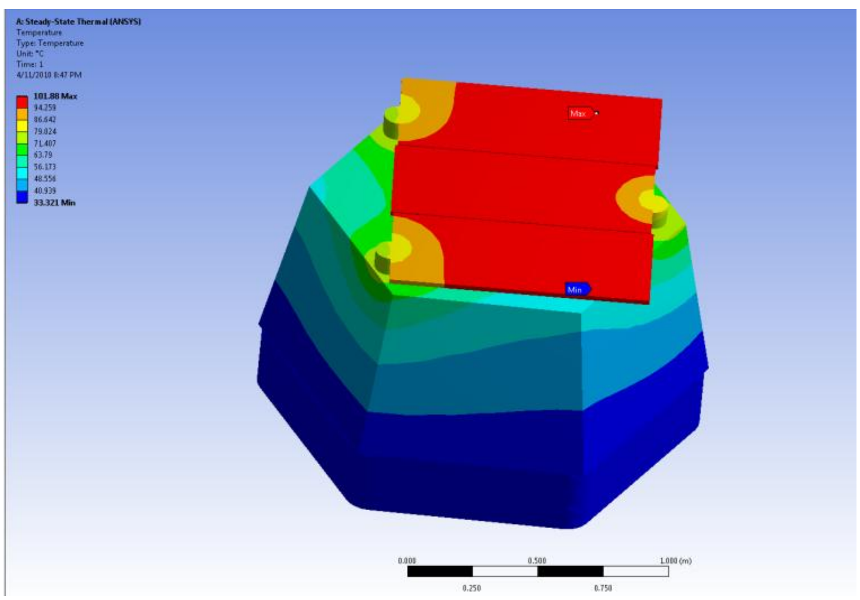
```
clc; clear all;
%% Thermal Simulation
% emis/absor matl: [AlKapton(1mil),BetaCloth,AlTeflon(5mil)]
disp('Matl: [AlKapton(1mil),BetaCloth,AlTeflon(5mil)] ')
 emisAL = [.67,.86,.78];
 absorpAL = [.38,.32,.16];
 IEarth = 231; % W/m^2 orbit average
 IMoon = 430; % W/m^2
 Isolar = 1367; % W/m^2 at ~ 1AU
 albedoE = .37;
 albedoM = .07; %in Lunar orbit
 periodE = 88.49*60; % s
 qsolar = 1367; % W/m^2
 Feir = .9401; % Flat plate approximation
 FalbedoE = .9486; % Flat plate approximation
 Fmir = 1; % Flat plate approximation
 FalbedoM = 1; % Flat plate approximation
 stef = 5.67e-8; %W/m^2-K^4

 %Spacecraft parameters
 emis = emisAL;
 absorp = absorpAL;
 Qint = 100; %W
  % Geometry
  % Cylinder
  % Projected area = 3.35m x 4.6m
  % A = 3.35*4.6;
  % Asurf = (3.35/2)^2*pi()*2+3.35*pi()*4.6;
  % Geometry
  %Actual Satellite
  %Area recieving flux
```

```
 A1 = 6.5*1.7; %Side facing
 modArea = (6*.42 + 6*.56 + 6*.2 + 6*.03);
 Asurf = modArea + 2.7/2*9 + 3*.42 + 2*pi()*4;
 A2 = 2*pi()*4; %top facing
%% Earth Orbit
 % Half time in sun, half in shade
 albedo = albedoE;
 qEarthIR = IEarth*Feir;
 qalbedo = absorp*Isolar*albedo*FalbedoE;
 qbackload = 0;

 % Maximum
 qext = qsolar + qalbedo + qEarthIR + qbackload;
 TmaxEarthOrbit = ((Qint + qext*A1)./(emis*stef*Asurf)).^(1/4) %in K
 % Minimum
 qext = qEarthIR + qbackload;
 TminEarthOrbit = ((Qint + qext*A1)./(emis*stef*Asurf)).^(1/4) %in K

%% In Transit
 % Assume albedo and EarthIR effects are negligible since
 % view factors past GEO are on the order of 1-2%

 qext = qsolar;
 Ttransit = ((Qint + qext*A1)./(emis*stef*Asurf)).^(1/4) %in K

%% Moon Orbit
 % Half time in sun, half in shade
 albedo = albedoM;
 qMoonIR = IMoon*Fmir;
 qalbedo = absorp*Isolar*albedo*FalbedoM;
 qbackload = 0;

 % Maximum
 qext = qsolar + qalbedo + qMoonIR + qbackload;
 TmaxMoonOrbit = ((Qint + qext*A2)./(emis*stef*Asurf)).^(1/4) %in K

 % Minimum
 qext = qMoonIR + qbackload;
 TminMoonOrbit = ((Qint + qext*A2)./(emis*stef*Asurf)).^(1/4) %in K
```

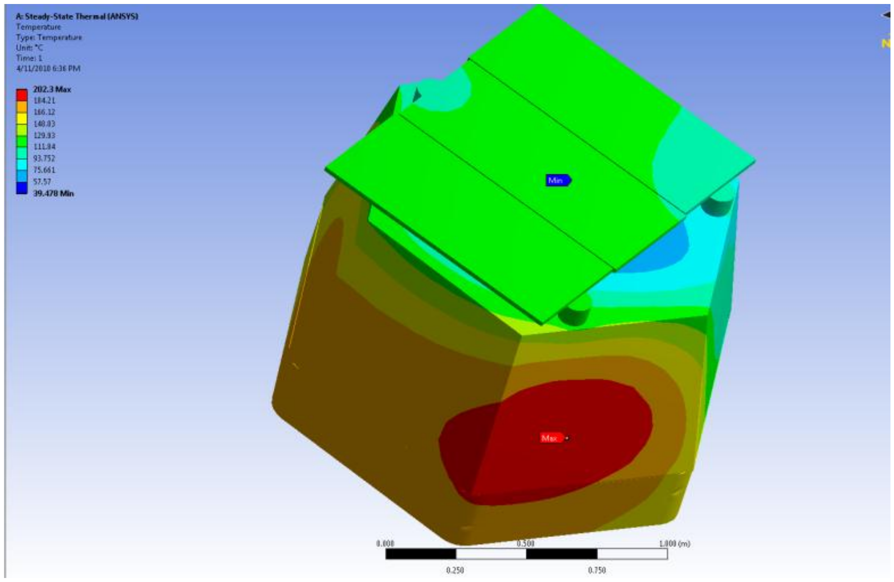
Thermal FEA:



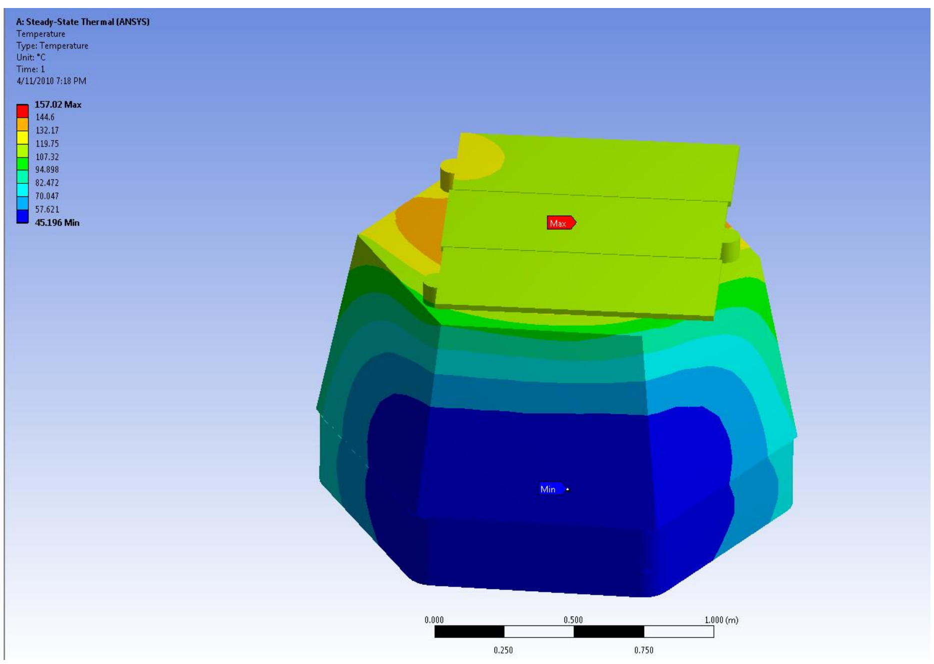
Appendix Figure 23: Earth Minimum Thermal Conditions



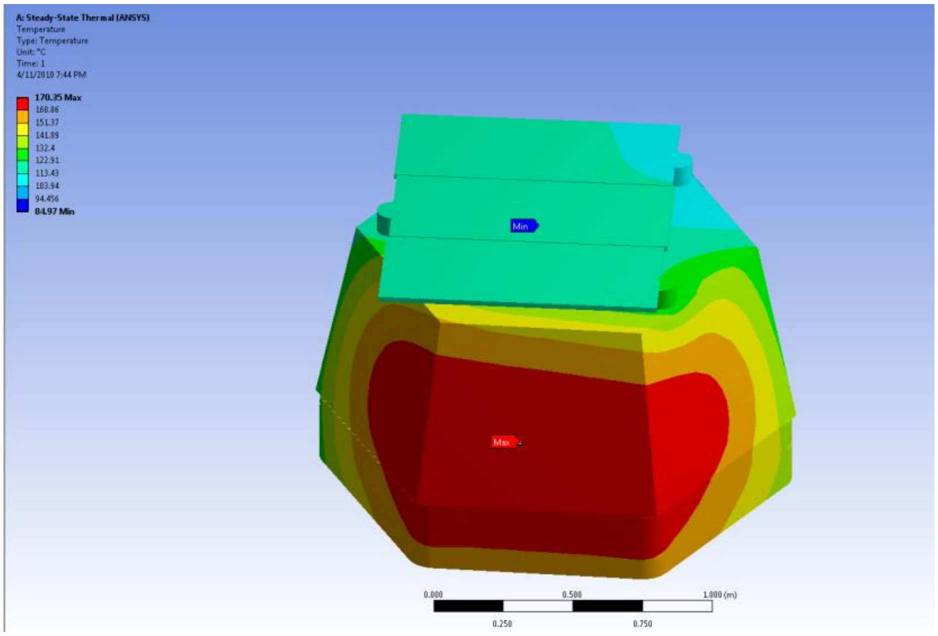
Appendix Figure 24: Average Transit Thermal Conditions, 30^{o} Lighting with Rotation



Appendix Figure 25: Moon Maximum Thermal Conditions Side Lit



Appendix Figure 26: Moon Maximum Thermal Conditions, 30⁰ Lighting



Appendix Figure 27: Moon Maximum Thermal Conditions, 60^{o} Lighting

	Maximum Temperature (C)	Minimum Temperature (C)	Internal gradient (C)
Earth Minimum	-73	-4	-4 to -50
Transit	102	33	33 throughout
Moon Maximum (worst case orientation)	202	39	76 to 160
Moon Maximum (30^{o})	157	45	
Moon Maximum (60^{o})	170	85	

Appendix Table 14: Thermal Analysis Results

Appendix N: Radiation Calculations

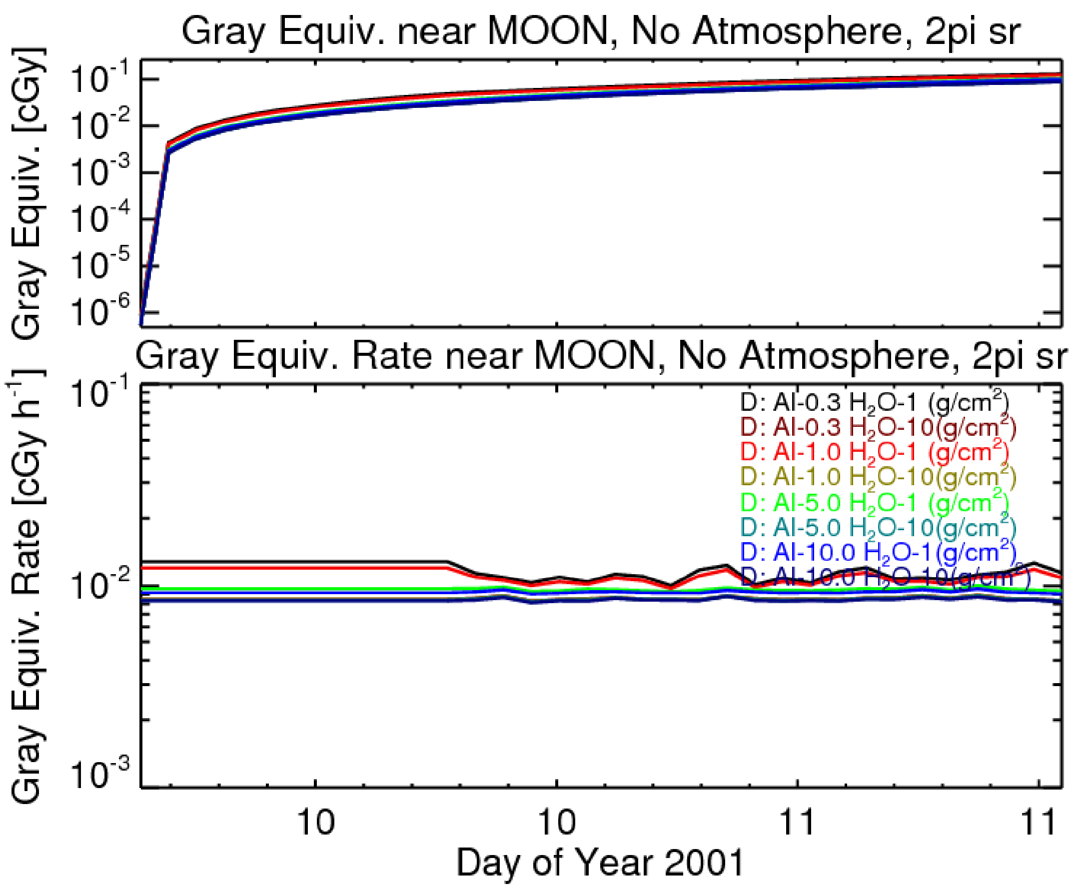
Solar winds are plasma flow emitted from the Sun. These are a constant source of radiation particle collisions compared to SPEs and CMEs. Typically, they are comprised of 2 categories: "fast wind" and "slow wind." The fast wind is a relatively new phenomena discovered by the Ulysses Spacecraft. It is difficult to characterize the fast wind component due to a large variance associated with properties (for instance, a 50% variance in flow speed was determined by Ulysses for fast wind compared to the 5% for slow wind). However, it is safe to assume that both the fast and slow winds are relatively similar at the Earth Moon system. The differences between these 2 components become more pronounced father away from the Sun.

The slow wind has been characterized by several satellite missions, although more data is needed to get a better statistical model beyond the Earth's magnetosphere. The composition of slow wind is approximately 92% H^+ and 8% He^{2+}, with trace amounts of C, N, O and Fe. Usually, the concentration of protons is 7.7 particles/cm^3. Each particle has a temperature of 91000K. However, because of the low concentration, the spacecraft macroscopically will experience much lower temperature. Of course, this depends on the location of the spacecraft. Finally the average flow speed of the plasma is about 400 km/s. Because of this, thermal coatings suffer extensive damage over the course of the mission. The thermal surfaces on MOM-E must survive 3 years worth of solar wind particle collisions. For about 74% of the Lunar orbit, the spacecraft will be in direct contact with the solar wind. Also worth mentioning is that solar wind intensity changes with the 11 year cycle that Sun goes through. The lowest activity in this cycle, called the solar minimum, corresponds to a less intense solar wind. Likewise, the highest intensity, solar maximum, corresponds to a higher intensity solar wind.

GCRs are the particles (i.e. a proton) that travel enter the solar system and are not emitted by the Sun or other radiating planets or objects of the solar system. The charged particles are constantly present in interplanetary space. They are independent of the time, direction, and location of the spacecraft. These high energy particles are the main source of Single Event Upsets (SEUs) in many data processing units for the spacecraft. SEUs refer to bit flips and other errors in data that high energy particles cause to data processing units. As such, extensive shielding must be in place around crucial electronics. This shielding involves putting layers of the base structural material (in this aluminum) around the components. GCR follow a 22-year cycle. The team will use GCR data from 1965 to 1970, which is representative a high GCR activity period in the 22-year cycle. Our baseline GCR intensity is around $4*10^4$ particles/m^2/s/sr. This is what Mom-E needs to withstand for the 3 year mission.

SPEs are rarer compared to GCRs and solar winds. Because of their sporadic occurrences, it is hard to predict how long each event lasts. There also large variations in intensity associated with each event. Most models limit themselves to near Earth orbits, and thus SPEs are harder to characterize for environments outside the magnetosphere.

The worst case phenomena a spacecraft can experience are CMEs. These are large ejections of plasma from the Sun's Corona that can sometimes cross paths with the Earth-Moon systems. It involves larger concentrations of very high energy particles relative to solar winds. However, because of their rarity, it is difficult to characterize a worst case baseline for CMEs. However, using the database collected from the SOHO spacecraft that extends as far back as 1996, the team can have a rough average to baseline the spacecraft radiation survivability criteria.



Appendix Figure 28: Sample EMMREM Output for Solar Minimum

Appendix O: Propulsion and Trajectory Calculations

Rocket	Payload to LEO (kg)	Fairing Size (m)	Country	Cost ($M)
Falcon-1	220	1.70	US	7.9
Soyuz-2	7,800	2.95	Russia	30-50
Proton-M	22,000	4.15	Russia	90
Atlas V	12,500	3.81	US	138
Delta II	5,089	2.44	US	60
Minotaur IV	640	1.67	US	12.5

Appendix Figure 29: Launch Vehicle Trade Study

The rocket selected was the Proton-M. Though the Proton is expensive and rather large, there were no smaller rockets found that could handle the necessary kickstage and payload mass. Because of the uniqueness of the mission, no rocket has been developed that is easily adaptable for the mission. Therefore, it was necessary to choose a rocket that provides more lifting power than is optimally necessary.

Kickstage	Available Vehicle	Delta-V (km/s)	Diameter (m)	Country	Cost ($M)
Briz-M	Proton-M	4.31	4.13	Russia	3
Centaur	Delta IV, Atlas V	9.55	2.05	US	20
Block DM-2M	Proton-M	4.60	3.7	Russia	4
Delta-K	Delta II	2.73	2.4	US	4.35
ATK Star 48V	Delta II	0.43	1.24	US	No Cost Found
Orion 38	Minotaur IV	0.57	0.97	US	2
Fregat	Soyuz-2	2.57	3.35	Russia	3-4

Appendix Figure 30: Kickstage Trade Study

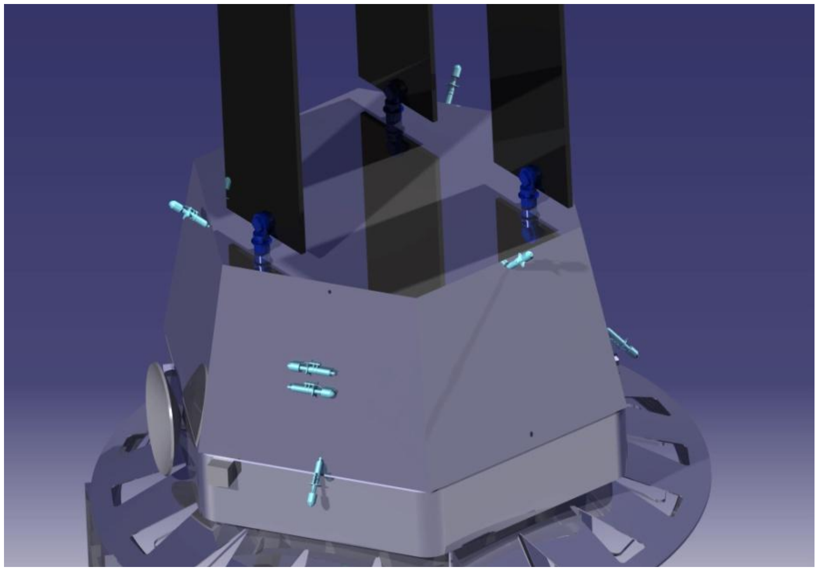
The kickstage selected was the Block DM-2M. This kickstage fits well into the Proton-M launch vehicle and provides more than the necessary ΔV for the MOM-E spacecraft to arrive in low Lunar Orbit. No other kickstage, with the exception of the Briz-M was a comparable choice. Longevity issues are the main factor for not selecting the Briz-M. If time and money allowed, a reconfiguration of the Briz-M could allow it to be used for this mission.

Transfer Orbit	Direct Elliptical Transfer	3.5 Phasing Loop Transfer	Weak Stability Boundary Transfer
Time of Flight (days)	5.2	22.18	90
Trans-Lunar Insertion (TLI) Delta-V (km/s)	3.137	3.096	3.194
Lunar Orbit Insertion (LOI) Delta-V (km/s)	0.816	0.873	0.712
Total Delta-V (km/s)	3.953	3.969	3.906

Appendix Figure 31: Transfer Orbit Trade Study

The selected transfer orbit must use as little propulsive power as possible and do the transfer in a timely manner. The three transfer orbits that were considered all had comparable ΔVs but had rather varying times. Obviously, the shortest time of flight is the one that was chosen.

Appendix P: Thruster Layout



Appendix Figure 32: Thruster Orientations

Appendix Q: Main Processor Trade Study

The Proton400k-L processor by Space Micro was chosen as the main board for the MOM-E mothership. The Proton 400k-L is shown in Appendix Figure 33.

Appendix Figure 33: Proton 400k-L Processor and Motherboard

This radiation-hardened processor is both power-efficient and powerful. The processor is a 1.2 GHz dual core based off the PowerPC architecture. It can process a total of 3600 million instructions per second. The entire board is rated to only draw between 8 and 12 watts of power. Of all the boards compared, this board also has the most options for different bus connections. This includes SPI, I^2C, CAN, UART, and others. The boards specifications indicate a RAM capacity up to 512MB.

For the C&DH architecture, a watchdog processor is also required. The watchdog processor looks at all network input for a reset command. If the processor sees the reset command, it sends a signal to restart the main processor. Broad Reach Engineering produces a processor board combination that will be suiTable for this task. The processor runs at a modest 133 Mhz, and the board draws 9 watts at peak. While this board will not be capable of the data compression and filtering that the main processor is capable of, this board should be able to simply relay information if anything damaging occurs to the main board. This board also supports 512 MB of RAM.

Appendix R: Data Bus Trade Study

Due to the high data rate requirements to maintain a 25 Mb/s link between the mothership and the ground station, a high-speed protocol is required onboard the mothership. The Itra-integrated Circuit(I2C), Serial Peripheral Interface (SPI), Spacewire, Controller Area Network (CAN), and Ethernet protocol were investigated as potential solutions to this issue.

The I2C protocol is a well know and well developed protocol used in spacecraft electronics. It consists of 2 wires hooked between a master chip and all of the slave chips. The master chip addresses the salve chip it chooses to get information from. The disadvantage of this protocol is the data throughput of the protocol. The High Speed mode for I2C can transfer 3.4 Mb/s maximum. Most hardware does not support I2C in high speed mode, so most systems can transfer only 400 kb/s.

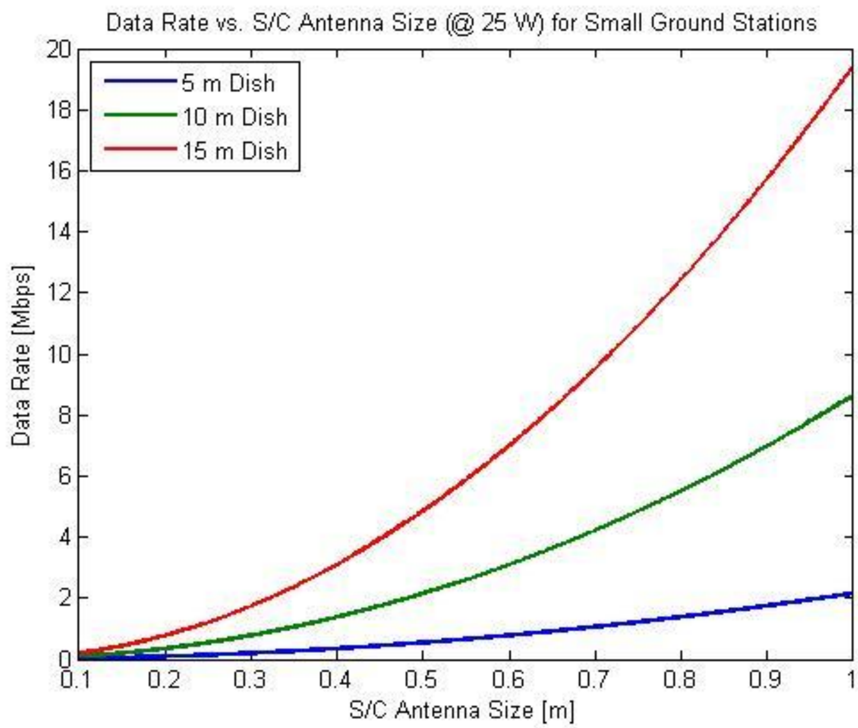
The SPI protocol utilizes a 4+ wire design. The wires are a Master In Slave Out (MISO) line, a Master Out Slave In (MOSI) line, a clock like and a chip select line. Each chip needs a chip select line as this is how the master chip addresses each slave. The data rate for this mode is dependent on the clock speed, but most sensors have a minimum response time that will limit this protocol. Also, the number of chip select lines necessary for all of the hardware would limit the kind for processor that could be used, as a large number of I/O ports would be used to maintain the protocol.

SpaceWire has a large heritage on spacecraft missions. The protocol used needs little software to maintain it and is fault tolerant. The protocol can achieve data rates from 2Mb/s to 400 Mb/s. Unfortunately, this protocol is not as prolific as other protocols and may be a deterrent to companies that would like to use the bus system.

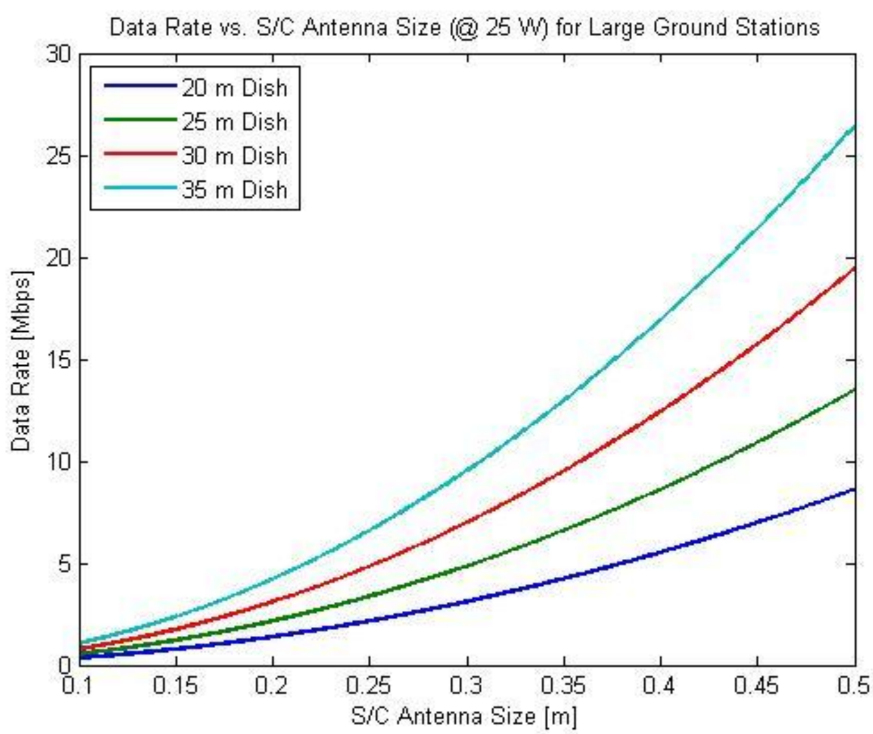
Ethernet has little heritage in space but has an extremely large user base on Earth. Some 802.1 Ethernet protocols are extremely fault tolerant and easy to interface with. Ethernet also is able to maintain data rates up to 100 Mb/s. The amount of software to maintain this protocol is larger but well tested and documented.

After finding the maximum data transfer rate for each protocol as well as looking at the user base for each, the 100base-TX standard Ethernet was chosen. This system uses the standard CAT-5 cabling to connect components to central hubs. The flight computer will also be connected directly to the Guidance, Navigation and Control (GNC) system via a SPI or I2C interface as well as the Communication system with Ethernet crossover cables.

Appendix S: Ground Station Analysis



Appendix Figure 34: Small Ground Station Analysis



Appendix Figure 35: Large Ground Station Analysis

Appendix T: Link Budgets

Item	Symbol	Units	Value
Frequency	f	Hz	8.00E+09
Transmitter Power (RF)	P	Watts	2.50E+01
Transmitter Power (RF)	P	dBW	1.40E+01
Transmitter Line Loss	L_l	dB	-2
Transmit Antenna Pointing Loss	L_{pt}	dB	-2
Propagation Path Length	S	m	3.84E+08
Space Loss	L_s	dB	-222.198
Propagation & Polarization Loss	L_a	dB	-1
Receive Antenna Diameter	D_r	m	34
Receive Antenna Efficiency	⍰$_r$	--	0.55
Receive Antenna Pointing Loss	L_{pr}	dB	0
Receive Antenna Gain (net)	G_r	dBi	66.505
System Noise Temperature	T_s	K	300
Data Rate	R	bps	1.30E+07
E_b/N_o (1)	E_b/N_o	dB	1.39E+01
Bit Error Rate	*BER*	--	1.00E-04
Transmit Antenna Diameter	D t	m	0.4
Transmit Antenna Efficiency	⍰t		0.55
Transmit Antenna Gain (net)	G_t	dBi	2.79E+01
Half Power Beamwidth	ϑ	°	6.5625

Appendix Table 15: Primary High Gain Antenna Link Budget

Item	Symbol	Units	Value
Frequency	f	Hz	8.00E+09
Transmitter Power (RF)	P	Watts	2.50E+01
Transmitter Power (RF)	P	dBW	1.40E+01
Transmitter Line Loss	L_l	dB	-2
Transmit Antenna Pointing Loss	L_{pt}	dB	-2
Propagation Path Length	S	m	3.84E+08
Space Loss	L_s	dB	-222.198
Propagation & Polarization Loss	L_a	dB	-1
Receive Antenna Diameter	D_r	m	34
Receive Antenna Efficiency	▯$_r$	--	0.55
Receive Antenna Pointing Loss	L_{pr}	dB	0
Receive Antenna Gain (net)	G_r	dBi	66.505
System Noise Temperature	T_s	K	300
Data Rate	R	bps	1.20E+05
E_b/N_o (1)	E_b/N_o	dB	1.23E+01
Bit Error Rate	BER	--	1.00E-04
Transmit Antenna Diameter	D t	m	N/A
Transmit Antenna Efficiency	▯t		0.55
Transmit Antenna Gain (net)	G_t	dBi	6.00E+00
Half Power Beamwidth	ϑ	°	110

Appendix Table 16: Secondary Low Gain Link Budget

Item	Symbol	Units	Value
Frequency	f	Hz	2.00E+09
Transmitter Power (RF)	P	Watts	1.10E+01
Transmitter Power (RF)	P	dBW	1.04E+01
Transmitter Line Loss	L_l	dB	-2
Transmit Antenna Pointing Loss	L_{pt}	dB	0
Propagation Path Length	S	m	2.13E+06
Space Loss	L_s	dB	-165.03
Propagation & Polarization Loss	L_a	dB	-1
Receive Antenna Diameter	D_r	m	0.65
Receive Antenna Efficiency	η_r	--	0.55
Receive Antenna Pointing Loss	L_{pr}	dB	0
Receive Antenna Gain (net)	G_r	dBi	6
System Noise Temperature	T_s	K	290
Data Rate	R	bps	4.00E+04
E_b/N_o (1)	E_b/N_o	dB	1.23E+01
Bit Error Rate	BER	--	1.00E-04
Transmit Antenna Diameter	D t	m	N/A
Transmit Antenna Efficiency	ηt		0.55
Transmit Antenna Gain (net)	G_t	dBi	6.00E+00
Half Power Beamwidth	ϑ	°	110

Appendix Table 17: Intersatellite Link Budget

Appendix U: Risk Analysis

ADCS

Likelihood					
5					
4					
3					
2	2	4,7,8,9			
1		10	1,5,6	3,	11,12,13
	1	2	3	4	5

Impact

Payload

Likelihood					
5					
4					
3					
2				5	
1		6	3,4,9,10,13,14	2,8,12	1,7,11
	1	2	3	4	5

Impact

Sructures

Likelihood					
5					
4					
3				2	
2				4,6,8,9	
1			5	3,7	1
	1	2	3	4	5

Impact

Comm

Likelihood					
5					
4	3				
3	10			9	
2		1,2,6	4,5	7	
1			8		
	1	2	3	4	5

Impact

Power

	5					
	4		1			
Likelihood	3		10		3	
	2		7		4,8	2,5,6, 9,13,14
	1			11		12
		1	2	3	4	5

Impact

C&DH

	5					
	4					
Likelihood	3			1		
	2			4	8	2
	1	7	3,5	6	9	
		1	2	3	4	5

Impact

Thermal/Rad

	5					
	4		5,7			
Likelihood	3	3				
	2		1			
	1				4	2,6
		1	2	3	4	5

Impact

Business

	5					
	4					
Likelihood	3			1	3	
	2				2	4
	1					
		1	2	3	4	5

Impact

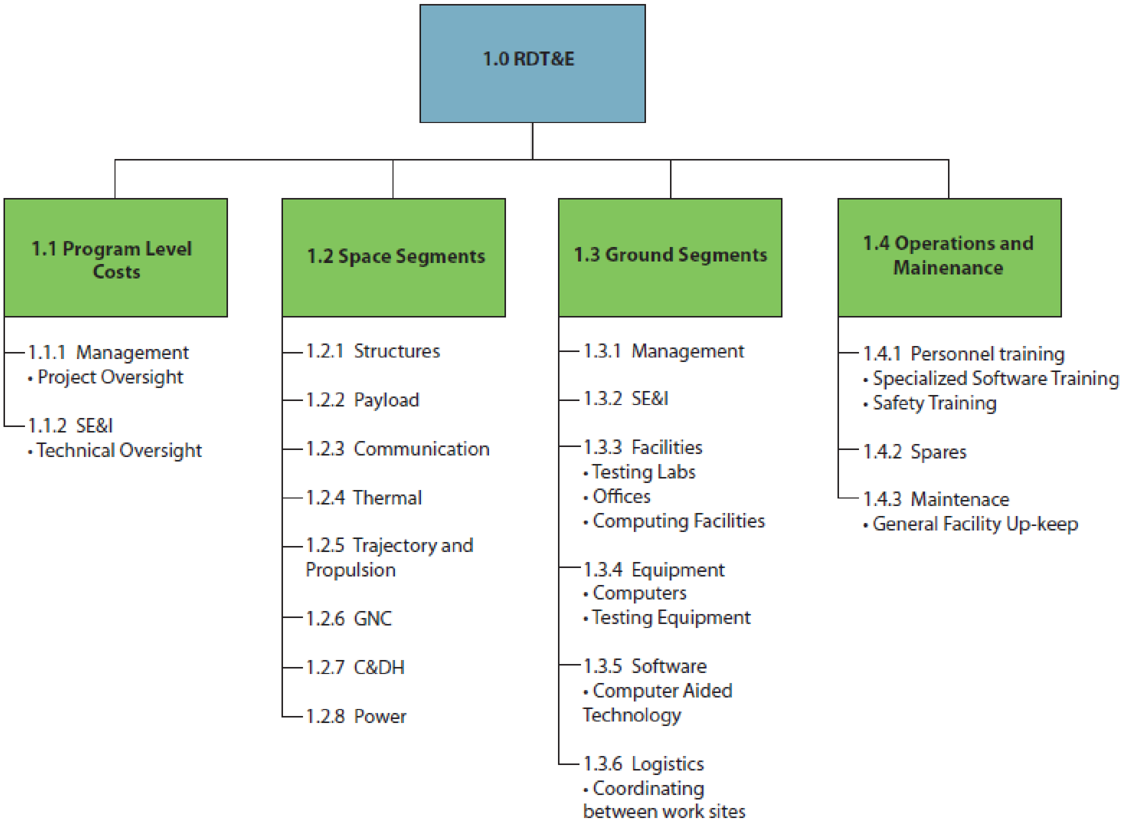
Risk ID	**High Risk Item**	**Description**	**Mitigation Plan**
ADCS-001	Kickstage ADCS Failure	After lunar transfer initialization, kickstage ADCS system cannot control vehicle	Spacecraft ADCS takes over for spin-stabilization en-route to the moon
ADCS-002	Minimal Thruster Failure	No more than one thruster in a given axis malfunctions	Multiple thrusters on each axis for redundancy; testing
ADCS-003	Critical Thruster Failure	More than one thruster in a given axis malfunctions simultaneously	Investigation of limited stability algorithms; testing
ADCS-004	Single Reaction Wheel Failure	One reaction wheel malfunctions	Fourth reaction wheel for redundancy on any axis; testing
ADCS-005	Multiple Reaction Wheel Failure	Multiple reaction wheels malfunction simultaneously	Extensive testing
ADCS-006	Momentum Unloading Failure	Thrusters cannot perform momentum-unloading procedures to relieve reaction wheels	Extensive testing
ADCS-007	IMU Failure	IMU cannot transmit sensing data to flight computer	Rely on star trackers and sun sensors for redundancy; testing
ADCS-008	Star Tracker Failure	Star tracker(s) cannot transmit data to flight computer	Rely on IMU and sun sensors for redundancy; testing
ADCS-009	Distorted Optics	Debris/gases/thermal issues cause defects in star tracking use or accuracy	Communication with thermal/radiation and orbits/propulsion subsystems
ADCS-010	Sun Sensor Failure	Sun sensor(s) cannot transmit data to flight computer	Rely on star trackers and IMU for redundancy; testing
ADCS-011	Full Sensor Failure	Multiple or all sensors fail	Redundant sensor types, extensive testing
ADCS-012	Full Actuator Failure	Both actuator systems fail simultaneously	Redundant actuator types, extensive testing
ADCS-013	Transponder Failure	Transponder is unable to relay flight data to Earth	Redundant transponders
PWR-001	Gimbal Lock	Gimbal lock	Stabilization of the Space Craft to ensure that solar panel gimbals do not lock
PWR-002	Gimbal Failure	Failure of gimbal	Properly test and power gimbal
PWR-003	Gimbal Load failure	Gimbal mechanism not able to withstand liftoff	Design system to help protect and support gimbal mechanism
PWR-004	Deployment	Deploying and restoring panels before kickstage	Develop support system for panels when they are brought back in to prevent damage when kickstage is fired
PWR-005	Deployment Communication	Failure to communicate to deployment mechanism for rails	Ensure that the battery, Frangibolt, and microcontroller have been properly tested and installed
PWR-006	Lightband Communication	Failure to communicate with Lightband releasing mechanism for payload	Ensure that the battery, Lightband, and microcontroller have been properly tested and installed
PWR-007	Health Communication	Failure to communicated health of payload to mothership computer	Ensure that proper cabling and a redundancy is in place

PWR-008	Radiation	Radiation degradation of communication lines	Ensure that communication lines are properly protected
PWR-009	Radiation	Irregular or unexpected radiation	Account for in solar panel sizing as an added contingency
PWR-010	Debris	Collision with debris or micrometeoroids	Account for in solar panel sizing as an added contingency
PWR-011	Cell Fabrication	Failure in cell fabrication	Account for in solar panel sizing as an added contingency
PWR-012	Power Distribution Board	Power distribution board failure	Test and ensure that board will not fail
PWR-013	Array Power Regulator	Array power regulation module failure	Test and add a contingency if failure occurs
PWR-014	Battery Charge Regulator	Battery charge/discharge regulation module fails	Test and add a contingency if failure occurs
BUS-001	Partially Empty Payload Bays	Not enough customers to cover production costs	Delay launches until bays are filled to maximum capacity
BUS-002	ITAR Red Tape	ITAR restrictions associated with United States technology launching in foreign territory	Hire a legal team to work around international laws
BUS-003	Limited Start-Up Funding	Venture capitalists do not find this venture a company worth investing in	Create a sound business plan that is marketable to a diverse type of venture capitalists around the world.
BUS-004	Limited Profit Margin	Constrained profit margin leaves little margin for manufacturing or other errors	Run a highly efficient business that can capitalize on the efficiency savings and turn it to profit.
PLD-001	Payload Separation Mechanical Failure	Bottom payload does not jettison	Use of highly robust separation device
PLD-002	Structure Hits Payload	Rings and bars do not eject normal to the craft and hit the payload	Testing
PLD-003	Health Check Failure	Signal is not received/processed by payload	Close integration with mothership
PLD-004	1st Separation Battery Failure	Lightband does not receive separation power	Redundant system
PLD-005	1st Payload Separation Signal Failure	Signal is not sent/received to/from Lightband to separate	Close integration with mothership
CDH-001	Data Storage Unit Failure	Component failure due to radiation exposure; still able to transmit data to Earth during downlink window, simply not able to store	Enforce unit with radiation hardened enclosure or material
CDH-002	Passive Flight Ethernet Splitter Failure	Malfunction of the physical splitter unit	Anchor splitter well
CDH-003	Passive Communications Ethernet Splitter Failure	Malfunction of the physical splitter unit	Anchor splitter well
CDH-004	Flight Switch Failure	Failure to route data between flight computer, payloads, and communications. Potential radiation failure	Backup switch installed. Both flight switch and backup switch in radiation hardened enclosure
CDH-005	Cable Failure	Radiation degradation of cable components	Run the cables through radiation hardened areas of the mothership
CDH-006	Wire Disconnection	Excessive vibrations or unforeseen movement of internal components causes disconnect	Reduce wire motion with staples

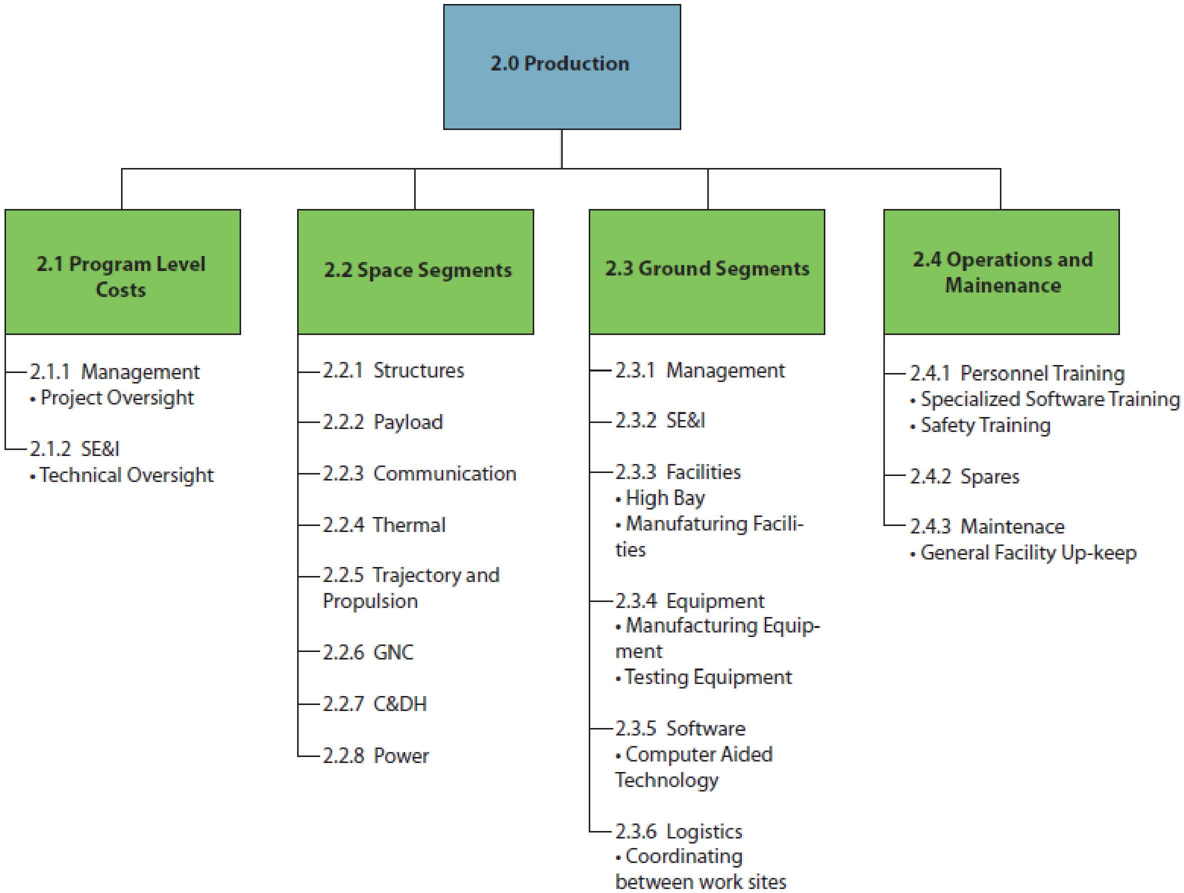
CDH-007	Interpayload Connection Failure	Interface communications between payloads malfunction	More wiring
CDH-008	Processor Failure - Hardware	Excess voltage events or failure to mitigate heat from unit cause failure	Processor is radiation hardened, space rated radiators
CDH-009	Processor Failure - Software	Infinite loop/segmentation fault	Watchdog processor, excellent code
STR-001	Structure failure to survive launch	Collapse of entire mothership	Optimize component to have SF > 1.2
STR-002	Railings Impeding Payload Deployment	Payload may be damaged on deployment or may not deploy	Deploy railings first to clear way for deployment of payload
STR-003	Payload/Kickstage Fails to Deploy; Spring/Lock Mechanism Not Releasing	Payloads behind this payload won't deploy	Make redundant deployment system so each payload has more of a chance to deploy
STR-004	Solar Panels Vibrational Damage During Launch or Kickstage Initiation	Reduced power to entire system; mothership won't be able to act as communications relay for payloads	Design strong interface between panels and mothership; Put panels far away from kickstage
STR-005	Collision with Space Debris	Damage to communications dish or solar panels; Increase the stress on the support rails	Design to withstand small collisions structurally or shield components
STR-006	Torsional Loading/Effects	Structural and payload damage	Design cross supports in truss design to reduce torsional effects
STR-007	Payload Interface to Mothership Fails	Detachment of payload from mothership; destruction of payload and/or other payloads	Stress analysis and plan for SF > 1.2
STR-008	Structure Experiences Resonant Frequency	Structural, component, and payload damage	Vibration testing to reveal resonant frequency
STR-009	Structure Experiences Side Loading	Rail structures around payload breaking; structure collapsing	Over anticipate loading in analysis; Design rails to best resist compressive loading
THM-001	Primary Heater Failure	The primary heaters fail	Secondary heaters take over
THM-002	Primary and Secondary Heater Failure	The secondary heaters fail	Run high power in eclipse
THM-003	Temp Sensor Fail	A temperature sensor fails	Other temperature sensors take data
THM-004	All Temp Sensors fail	All of the temperature sensors fail	Turn heaters on and off based on previous orbits
THM-005	Insulation Issue	Some rips the insulation or it becomes detached	Heat more in eclipse; limit time that damaged section is in sunlight
THM-006	Big Insulation Issue	Total loss of insulation	Heat more in eclipse; do not produce heat (run low power) in sun
THM-007	Radiation Shield Issue	Damage to shield due to micrometeorite or surface discharge	Components will receive more radiation contingency should handle this

COM-01	Switch Failure	Switches between antennas and radios fail to flip	Two transceiver units for redundancy, testing
COM-02	X-Band Radio Failure	Radiation/other source causes X-band radio to fail	Two transceiver units for redundancy, testing
COM-03	Single S-Band Radio Failure	Radiation/other source causes S-band radio (receiver OR transceiver) to fail	Two transceiver units for redundancy, testing
COM-04	Transceiver Unit S-Band Failure	Radiation/other source causes S-Band receiver and transceiver to fail in one transceiver unit	Two transceiver units for redundancy, testing
COM-05	Antenna Damage During Launch	HGA or LGAs jostled and damaged by launch vehicle during launch and misshapen; radiation pattern altered	Vibration testing, contingency in fairing dimensions
COM-06	Interference/Reflections	Placement of antennas causes interference of radiation patterns; other components may reflect signal	Testing
COM-07	Wiring/Coax Harness Failure	Coax and other wiring dislodged during launch; connections from radios to antennas broken	Vibration testing, structural analysis
COM-08	Micrometeorite Damage	Micrometeorite strikes either HGA or LGA during mission operations, deforms antenna; radiation pattern altered	Investigate antenna material that can withstand impact
COM-09	Dual-Band Bandwidth Insufficient	LGA dual-band capability may be insufficient for payload communication or may delay schedule in design	Anechoic chamber testing, antenna re-selection; this would lead to moderate system redesign
COM-10	Micro-Patch Array Development	Development of micro-patch array proves infeasible or delays development significantly	Cut micro-patch array from design

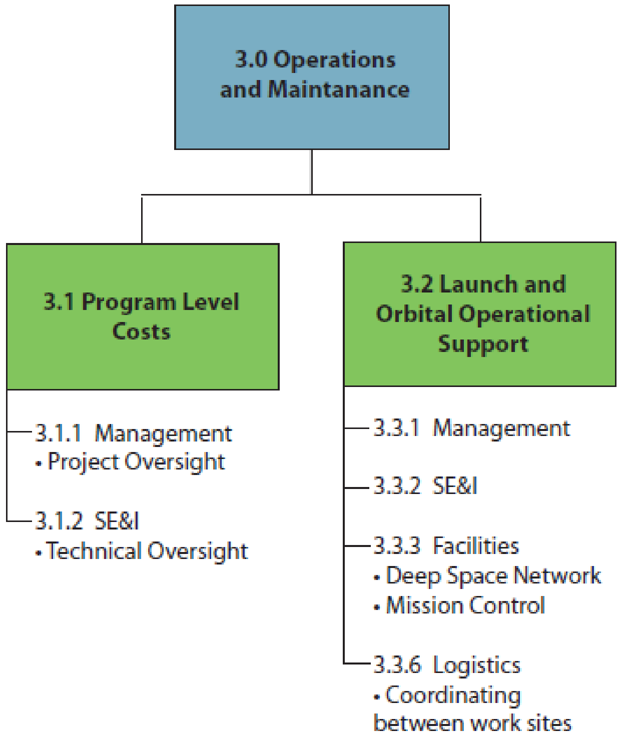
Appendix V: Business Models



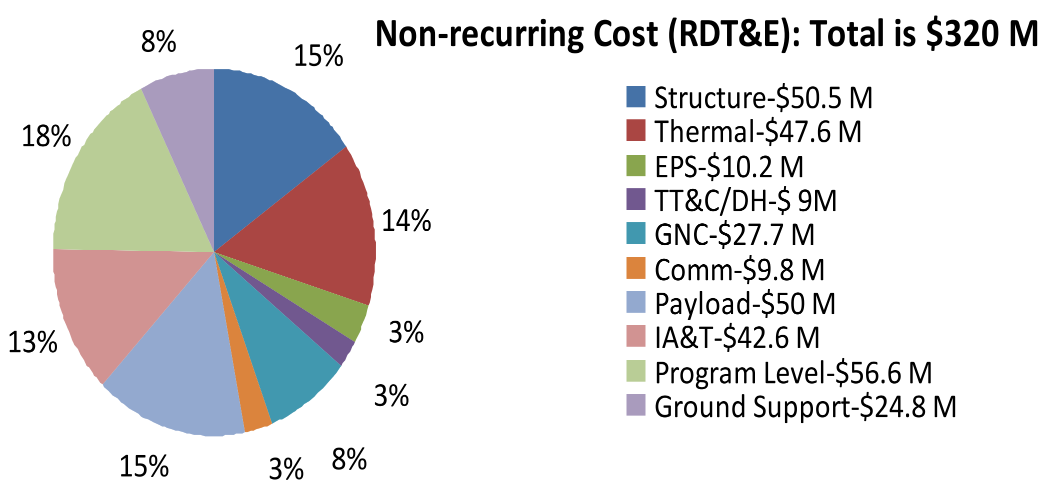
Appendix Figure 36: Breakdown for Research, Design, Test, and Evaluation Work Breakdown



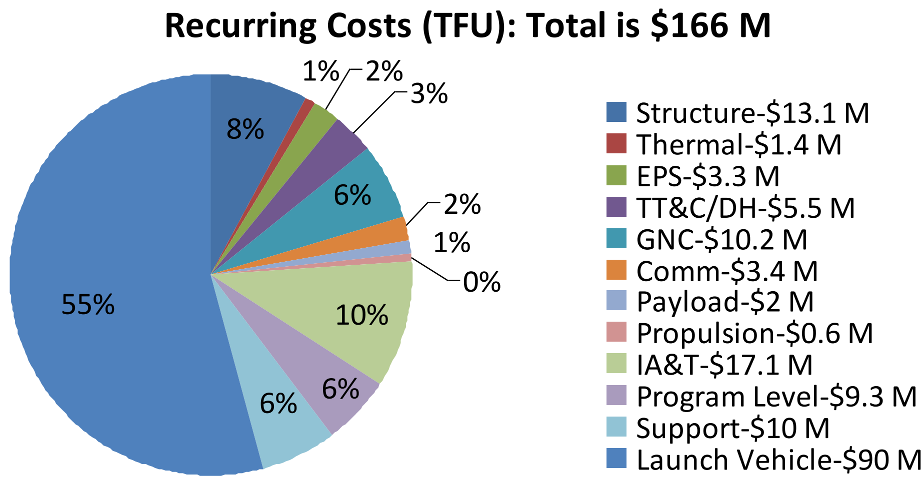
Appendix Figure 37: Production Phase Work Breakdown Structure



Appendix Figure 38: Operations and Maintenance Work Breakdown Structure



Appendix Figure 39: Non-Recurring Cost Breakdown



Appendix Figure 40: Recurring Cost Breakdown

$$Cost\ for\ Customer\ (\$) = TPC \cdot \frac{\left[\frac{m_C}{M_T}+\frac{h_C}{H_T}\right]}{2}$$

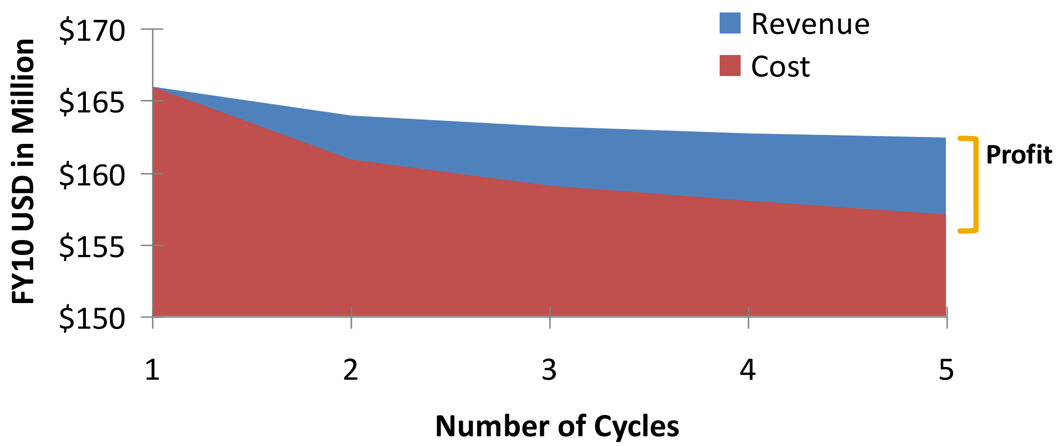
Where:

- TPC = total production cost
- m_c = mass of the individual customer
- M_T = total mass of all customers
- h_c = height of customer
- H_T = total height of all customers

Appendix Figure 41: Cost Distribution Equation

Customer Type	Mass (kg)	Height (m)
Large	500	1.3
Medium	350	1
Small	150	0.6
Instrument	100	0.5
CubeSat (30 “U” Max)	300	0.6

Appendix Figure 42: Standardization of Mass and Height for Each Customer Type



Appendix Figure 43: Profit Margin for First 5 Mission Cycles